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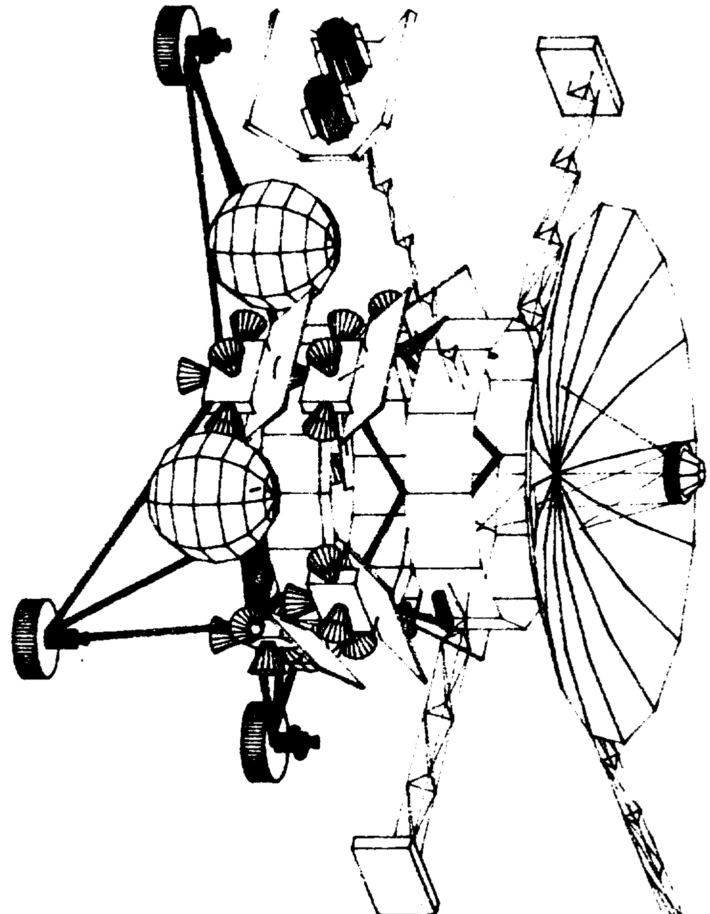
Asteroid Exploration and Utilization



The Hawking Explorer

Submitted to:

NASA/USRA

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Presented by:

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Executive Overview

The Earth is nearing depletion of its natural resources at a time when human beings are rapidly expanding the frontiers of space. The resources which may exist on asteroids could have enormous potential for aiding and enhancing human space exploration as well as life on Earth. With the possibly limitless opportunities that exist, it is clear that asteroids are the next step for human existence in space.

This report comprises the efforts of NEW WORLDS, Inc. to develop a comprehensive design for an asteroid exploration/sample return mission. This mission is a precursor to proof-of-concept missions that will investigate the validity of mining and material processing on an asteroid.

Project STONER (Systematic Transfer of Near Earth Resources) is based on two utilization scenarios:

- Moving an asteroid to an advantageous location for use by Earth.
- Mining an asteroid and transporting raw materials back to Earth.

The asteroid explorer/sample return mission is designed in the context of both scenarios and is the first phase of a long range plan for humans to utilize asteroid resources.

The report concentrates specifically on the selection of the most promising asteroids for exploration and the development of an exploration scenario. Future utilization as well as subsystem requirements of an asteroid sample return probe are also addressed.

Project STONER is divided into two primary areas: asteroid selection/mission design and explorer spacecraft design. The asteroid selection team has narrowed the possible 4800+ known asteroids to ten, considering physical attributes of each candidate asteroid as well as mission trajectory and ΔV requirements. From that group of ten, a final asteroid was chosen for more in-depth study.

The mission design team formulated mission scenarios and -- working with the other teams -- investigated possible problem areas and contingency plans. In the design of the spacecraft, subsystems that have been studied are: GNC, communications, automation, propulsion, power, structures, thermal systems, scientific instruments, and mechanical retrieval devices.

The Hawking spacecraft, designed to study an asteroid and return a sample to Earth, was named after Steven F. Hawking as a tribute to his continuing efforts to expand the limits of man's understanding of the universe. The Hawking is an adaptation of the Mariner Mk II series of spacecraft. Utilization of the Mariner Mk II design can accelerate development of the spacecraft and significantly reduce cost.

The Hawking spacecraft consists of three component vehicles: orbiter, lander and the sample return craft (SRC). Each of these vehicles has specific mission objectives and contributes directly to the fulfillment of the primary mission goal: return a sample of asteroidal material to the earth for analysis. Analysis of the samples is crucial in determining the composition of different taxonomic classes, and is a necessary step before utilization of asteroids can begin.

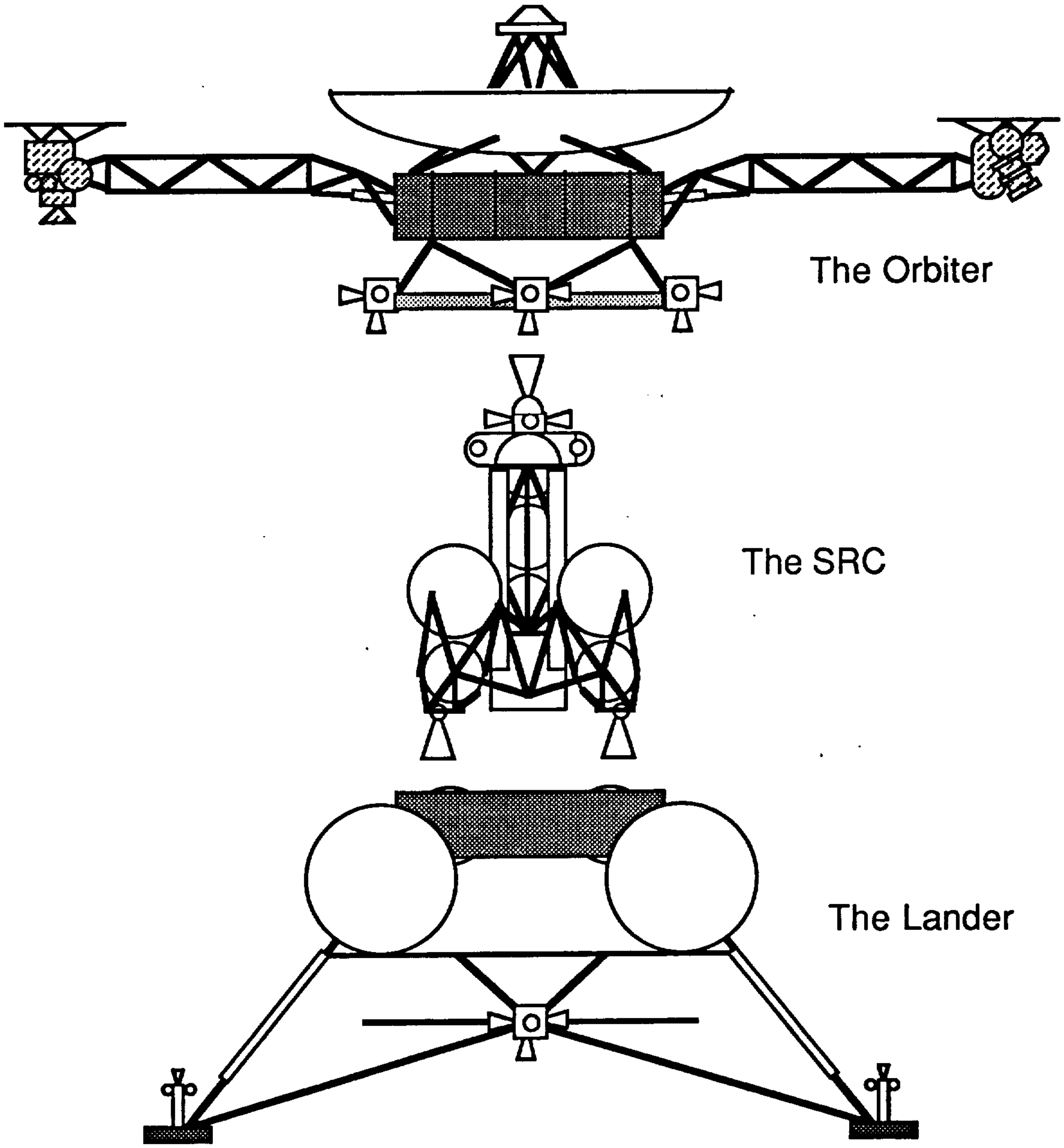
The sample return mission scenario consists of nine phases, the successful completion of each being critical to the overall mission:

Phase 1: Hawking is launched into LEO orbit aboard an existing launch vehicle.

Phase 2: Hawking is injected into the interplanetary transfer trajectory by its upper stage.

Phase 3: During the interplanetary cruise, the spacecraft performs radio science experiments and studies of the solar wind.

Phase 4: Hawking inserts itself into the asteroid's orbit, positioning itself several asteroid radii ahead of the body and slightly to the sun side. This position allows the spacecraft to map the asteroid, determine its rotational axis, and locate scientifically interesting features, all to help determine a desirable landing site.



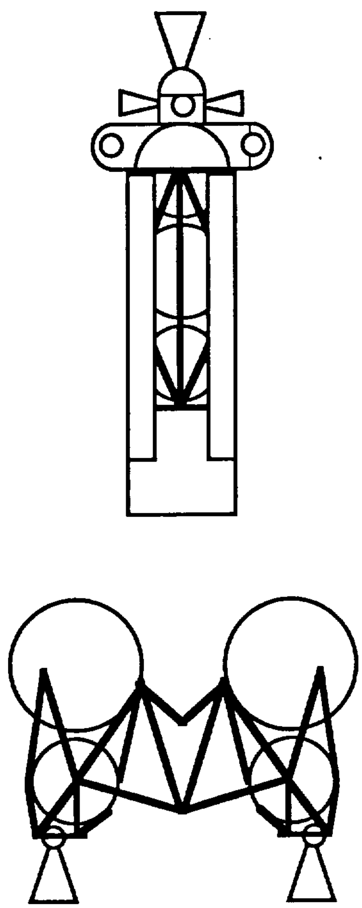
The Hawking Explorer Spacecraft

Phase 5: Once a landing site is chosen, the lander/SRC separates from the orbiter. The orbiter remains several asteroid radii away to serve as a relay for the lander/SRC and to provide reconnaissance for the rovers. The lander/SRC approaches and docks with the asteroid using its attitude control thrusters.

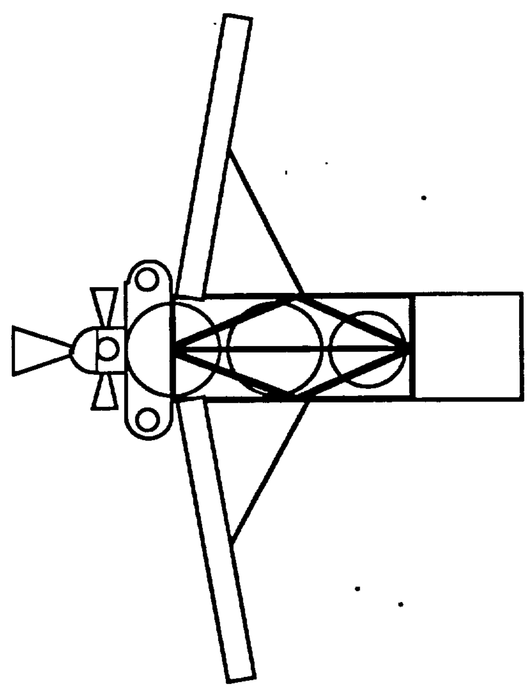
Phase 6: Samples of scientific interest are identified and retrieved by either the robotic arm or the rovers, and placed in the SRC.

Phase 7: The SRC is launched from the lander in a non-destructive manner (ie., springs, pneumatic pistons) so the lander can remain intact to perform more analysis of the asteroid. At an altitude of approximately 0.5 km above the lander, the SRC rotates and fires the booster's engines to inject itself into the transfer trajectory back to Earth.

Phase 8: After the injection burn, the booster stage is jettisoned and the communication antenna and solar panels are deployed. During the interplanetary cruise of the SRC, the integrity of the sample is maintained by minimizing g-loads during maneuvers and keeping the sample at a low temperature.



SRC Boosters are Jettisoned



SRC Panels are Deployed

Phase 9: Upon arrival at Earth, the SRC inserts itself into a highly elliptical orbit. After accurate ground-based orbit determination, the SRC circularizes into

LEO where it waits for pick-up by either the Space Shuttle or Space Station Freedom.

There are many benefits to be gained from studying asteroids. Presently these bodies have been almost totally neglected in the exploration of the solar system. It is believed that because of their primitive state they hold clues to the formation of the solar system. Also, utilization of asteroids as a future space-based source of raw materials could reduce the total cost of future space missions.

Acknowledgements

This report is the product of countless hours of researching, designing, re-designing, and reviewing by a group of eleven Aerospace Engineering students who decided to tackle a new project instead of redoing an old one.

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Table of Contents

Executive Overview....................i
Acknowledgements....................v
Table of Contents....................vi
List of Tables....................viii
List of Figures....................ix
List of Acronyms....................x

1.0 Background....................1
- 1.1 Depletion of the Earth's Resources....................1
- 1.2 Cost of Launching to LEO....................2
- 1.3 Scientific Value....................3

2.0 Asteroid Utilization....................4
- 2.1 Scenario I: Asteroid Retrieval....................4
- 2.2 Scenario II: Asteroid Mining/Refuelling Waystation....................4
- 2.3 Selecting Asteroids for Utilization....................4
  - 2.3.1 Overall Asteroid Selection Criteria....................5
  - 2.3.2 Mission Specific Selection Criteria....................5
- 2.4 Asteroid Selection Process....................6
- 2.5 Asteroid 3677....................7

3.0 Asteroid Exploration, the First Step....................9
- 3.1 Mission Objectives....................9
- 3.2 Mission Assumptions....................10
- 3.3 Layout of the Hawking Explorer....................10
- 3.4 Exploration Scenario....................12
- 3.5 Mission Scenario to Asteroid 3677....................18

4.0 The Hawking Explorer Design Philosophy....................22
- 4.1 Control System Integration....................23
- 4.2 Propulsion System Integration....................23
  - 4.2.1 Supporting Assumptions....................24
  - 4.2.2 Hawking Propulsion Scenario....................25
  - 4.2.3 Propellants....................29
  - 4.2.4 Propellant Tanks....................30
  - 4.2.5 Material Requirements....................31
  - 4.2.6 Hawking Mass Breakdowns....................32
- 4.4 Launch Systems....................33
  - 4.4.1 Upper Stage Selection....................33
  - 4.4.2 Launch Vehicle Selection....................34

5.0 The Orbiter....................36
- 5.1 Orbiter Mission Objectives....................36
- 5.2 Scientific Instruments....................36
- 5.3 Power....................37
- 5.4 Communications....................37

5.4.1 Communication Through the Phases....38
5.4.2 High Gain Antenna....38
5.4.3 Low Gain Antenna....40
5.5 Guidance, Navigation and Control....41
5.5.1 Orbiter GNC Requirements for Mission Phases....41
5.5.2 Spacecraft Stabilization....42
5.5.3 Sensors and GNC Actuators....42
5.6 Structures and Thermal Control....43
5.7 Orbiter Mass Allocation....45

6.0 The Lander....46
6.1 Lander Mission Objectives....46
6.2 Scientific Instruments....46
6.3 Power....47
6.4 Communications....48
6.5 Structures and Thermal Control....48
6.6 Lander Mass Allocation....48

7.0 The Sample Return Craft....50
7.1 SRC Mission Objectives....50
7.2 Power....50
7.3 Communications....51
7.4 Guidance, Navigation and Control....53
7.4.1 Spacecraft Stabilization....53
7.4.2 Actuators and Sensors....54
7.4.3 GNC Thrusters....55
7.5 Structures and Thermal Control....55
7.6 SRC Mass Allocation....56

8.0 Contingency Plans....57

9.0 Recommendations for Further Mission Development....59

10.0 Organization....60
10.1 Organization Structure....60
10.2 Integration....60
10.3 Responsibilities of Each Team....62
10.4 Cost of this Project....63

11.0 References....64

12.0 Bibliography....66

Appendix A: Asteroid Utilization....71
Appendix B: FORTRAN Optimization Code....84
Appendix C: ROCKET2 Program....90
Appendix D: Communication Link Design....105
Appendix E: CAD Drawings....106

List of Tables

Table 1. Some Mineral Commodities whose Reserves Expire. 1
Table 2. Scenario Specific Selection Criteria.. 6
Table 3. The Eight Asteroids Selected by ASTOUT.FOR. 7
Table 4. Information on the Mission to Asteroid 3677. 19
Table 5. Propulsion Requirements for a Hawking Mission to Asteroid.......... 25
Table 6. Propellant Characteristics.. 30
Table 7. Some Materials and their Properties.. 32
Table 8. Masses of Hawking's Component Vehicles....................................... 33
Table 9. Upper Stage Boosters.. 34
Table 10. Possible Launch Systems.. 34
Table 11. Orbiter Instruments... 37
Table 12. Parameters for the Orbiter High Gain Antenna.............................. 39
Table 13. Parameters for the Orbiter Low Gain Relay Antenna..................... 41
Table 14. Control Actuators 43
Table 15. GNC Sensors 43
Table 16. Operating Temperature Range. 44
Table 17. Mass Allocation for the Orbiter. 45
Table 18. Lander's Scientific Instruments and their Function........................ 47
Table 19. Mass Allocation for Lander Subsystems. 49
Table 20. Parameters for the SRC Low Gain Antenna 51
Table 21. Bias Momentum Stabilization Method 53
Table 22. Control Actuators 54
Table 23. GNC Sensors for the SRC.. 54
Table 24. Mass Allocation for the SRC Subsystems. 56
Table 25. Contingency Plans During Various Phases of the Mission. 57
Table 26. Team Responsibilities.. 62

List of Figures

Figure 1. Launch Window for 3677....8
Figure 2. Return Window for 3677....8
Figure 3. Exploded View of the Hawking Explorer....11
Figure 4. Hawking and Upper Stages Inside a Titan IV Shroud....12
Figure 5. Hawking Explorer in Cruise Configuration (Phase 3)....13
Figure 6. Orbiter and Lander/SRC Sections Separate (Phase 5)....14
Figure 7. Lander/SRC Configuration on the Asteroid Surface (Phase 6)....15
Figure 8. Non-Destructive Launch of the SRC from the Lander (Phase 7)....16
Figure 9. SRC Jettisons Boosters and Deploys Solar Panels (Phase 8)....17
Figure 10. Outbound Trajectory to Asteroid 3677 1982DB....20
Figure 11. Return Trajectory from Asteroid 3677 1982DB....21
Figure 12. Schematic of Hawking Propulsion System for Phases 3 & 4....26
Figure 13. Schematic of the Orbiter and lander/SRC Propulsion Systems....27
Figure 14. Schematic of the Booster and SRC Propulsion System....28
Figure 15. Schematic of the SRC Propulsion System during Phases 8 & 9....29
Figure 16. Orbiter Relay Communications Architecture....38
Figure 17. Change in Orbiter HGA Data Rate....40
Figure 18. Variation in Orbiter HGA Transmission Space Loss....40

List of Acronyms

AI	Artificial Intelligence
ALAS	Advanced Liquid Axial Stage
AME	Automated Mining Equipment
APU	Asteroid Propulsion Unit
CCD	Charged Couple Device
CLL	Common Lunar Lander
CRAF	Comet Rendezvous Asteroid Flyby
CW	Clohessy-Willshire
DSN	Deep Space Network
EML	Electromagnetic Launcher
EVA	Extravehicular Activity
FPTNK	Fuel Pressurant Tank
GNC	Guidance, Navigation and Control
HGA	High Gain Antenna
HLLV	Heavy Lift Launch Vehicle
HST	Hubble Space Telescope
ITV	Interplanetary Transfer Vehicle
IUS	Inertial Upper Stage
JPL	Jet Propulsion Laboratory
LEO	Low Earth Orbit
LGA	Low Gain Antenna
LH	Liquid Hydrogen
LOX	Liquid Oxygen
MCB	Midcourse Correction Burn
MPD	Magnetoplasmadynamic
NASA	National Aeronautics and Space Administration
NEA	Near Earth Asteroid
NPP	Nuclear Pulsed Propulsion
OPTNK	Oxidizer Pressurant Tank
PIT	Pulsed Inductive Thruster
RF	Radio Frequency
RTG	Radioisotope Thermoelectric Generator
SDI	Strategic Defense Initiative
SEI	Space Exploration Initiative
SRC	Sample Return Craft
SRM	Solid Rocket Motor
SRMU	Solid Rocket Motor Upgrade
STONER	Systematic Transfer of Near Earth Resources
STS	Space Transportation System
TCM	Trajectory Correction Maneuver
TOF	Time Of Flight
TOS	Transfer Orbit Stage
USRA	Universities Space Research Association
UV	Ultra-violet

1.0 Background

There are an estimated 5000 near Earth asteroids, all of which could contain *valuable* untapped resources. These asteroids are considered valuable because:

- The Earth's resources are depletable.
- The cost of launching to low Earth orbit (LEO) is high.
- The asteroids hold clues about the formation of the solar system.

NEW WORLDS, Inc. is investigating methods in which the natural resources present in asteroids can be utilized to benefit man both on the Earth and in space. The following sections provide justification for the utilization of asteroids.

1.1 Depletion of the Earth's Resources

The resources of the Earth are not infinite, in fact, some are depletable within the next several decades. Table 1 shows the projected depletion year of high grade ores.

Table 1: Some Mineral Commodities whose Reserves Expire in the Near Future (1)

Commodity	Projected Depletion (yr)
Antimony	2026
Barite	2018
Bismuth	2021
Copper	2017
Flourite	2003
Lead	2014
Mercury	2028
Silver	1999
Tungsten	2009
Zinc	2015

* Duration of reserves and conditional resources at 1971 consumption rates

Once high grade reserves are used up, low grades ores will have to be used. The effects of using low grade ores on the environment and society were discussed in the article "Mining Outer Space" by Dr. Thomas McCord of M.I.T. and Michael Gaffey of the University of Miami:

> As the recoverable [commodity] content of the crude ore deceases, more material must be excavated, more energy must be used to concentrate the [commodity] bearing phases, and more waste material must be disposed... the environmental costs *or* the financial costs of negating the environmental damage increase significantly as the grade of the crude ore decreases. These costs must be paid in a lowering of the quality of life *or* an increase in the cost of the materials *or* both(2).

Therefore, for society to continue to thrive and grow, the Earth's mineral and energy resources must be augmented by the resources available from the solar system.

1.2 Cost of Launching to LEO

Presently, the cost of launching from Earth into LEO on an available launch system averages approximately $8300/kg (3). While this cost is acceptable for small compact spacecraft (communication, exploration, etc.), it becomes burdensome for large, bulky space structures such as the upcoming Space Station Freedom. Even with the advent of a heavy lift launch vehicle (HLLV), launch costs per kilogram are unlikely to decrease since these advance concepts still use chemical propulsion.(4)

Asteroid resources in orbit could be used to construct the parts of large space structures such as trusses, plates, beams, habitation modules and perhaps solar arrays. Once the initial costs of development are overcome and the industrial infrastructure in place, the cost of processed material in orbit could approach those on Earth.

1.3 Scientific Value

It is believed that asteroids contain clues about the formation of the solar system that were erased on the larger planetary bodies. The relatively small size of asteroids excluded them from experiencing the evolutionary processes that altered the original structure of the planets and large moons. Therefore, it is suspected that asteroids are composed of primitive matter typical of the region in which they formed. Also, asteroids may contain evidence of long-term fluctuations in the solar wind, micrometeors, solar-flare particles, and galactic cosmic rays. The thorough study of asteroids could increase the knowledge of the birth and evolution of the solar system[(4)].

However, the most beneficial scientific return of an exploration mission may be the mineralogical classification of asteroids. Presently, scientists divide asteroids into several taxonomic classes (ie.: A, C, S, M, etc.) dependent upon the type of light they reflect. Although scientists attempt to extrapolate compositions for each category from this data, the actual mineralogical composition of asteroids is not clear. The opportunity to determine the mineralogical composition of several asteroids could aid in determining the composition of the various classes[(5)].

2.0 Asteroid Utilization

There are several ways to utilize the resources available from asteroids, all of which involve mining. Asteroids contain metals, silicates, volatiles and perhaps radioactive materials. The metals could be mined to augment depleted Earth resources or to construct large space structures. Silicates could be used to make large solar arrays while volatiles, such as water, could be used to refuel and resupply interplanetary ships. Radioactive materials mined from asteroids could fuel nuclear reactors or engines used in space. Therefore, these hazardous materials would not have to be lifted from Earth.

Two asteroid utilization scenarios have been developed by NEW WORLDS, Inc., and are discussed extensively in Appendix A. A brief overview of the two scenarios is presented below.

2.1 Scenario I: Asteroid Retrieval

In this scenario, a small near Earth asteroid (NEA) is transferred to an Earth orbit. The final orbit is dependent upon whether the mined materials are used to support terrestrial activities or for construction of space structures.

2.2 Scenario II: Asteroid Mining/Refuelling Waystation

In the second scenario, the asteroid is mined in its original orbit for both metals and volatiles. Some commodities are taken to the desired location (Earth, Moon or Mars) while others are used *in situ* by large interplanetary spacecraft (refueling station).

2.3 Selecting Asteroids for Utilization

The common aspect between the two utilization scenarios is an initial phase of *exploration* to discover the overall properties of asteroids. Thus, the main objective of exploration is to determine the physical and mineralogical properties of the major taxonomic classes of asteroids. NEW WORLDS, Inc. estimates five to ten sample return missions to different asteroids are necessary to obtain a representative cross-section of these classes.

To select candidates from the 4800+ known asteroids, selection criteria were developed that reflected both the overall objective of exploration and the objectives of each utilization scenario.

2.3.1 Overall Asteroid Selection Criteria

These selection criteria satisfy the scientific requirements of the exploration phase of the mission:

- All major taxonomic classes should be represented in the final selection group.
- Since a majority of asteroids are classified as S or C, the final group should contain at least two representatives from each of these classifications.
- The ΔV necessary to perform the mission should be reasonable.
- The length and frequency of launch windows should be reasonable.

2.3.2 Mission Specific Selection Criteria

The mission specific criteria are based on the particular utilization scenario. Since the two utilization scenarios are different, the asteroid selection criteria for Scenario I (Asteroid Retrieval) differs from Scenario II (Mining/Refueling Waystation). The mission specific selection criteria are shown in Table 2.

Scenario I requires a small asteroid (<1 km) because of propulsion constraints, while Scenario II requires a larger asteroid (> 1 km) to simplify mining operations and sustain utilization for a longer period.

Both scenarios utilize near Earth asteroids. Scenario I favors Earth crossing asteroids, most of which are M type, while Scenario II favors Earth approaching asteroids, most of which are C or S types.

Table 2: Scenario Specific Selection Criteria.

Scenario	Diameter (km)	Taxonomic Classification	ΔV (km/s)
Asteroid Retrieval	< 1.0	M	< 7.5
Asteroid Mining/ Refueling Waystation	> 1.0	S or C	< 8.0

The last criterion is the ΔV restriction. For the Asteroid Retrieval Scenario, the total ΔV for an outbound mission to the asteroid should be less than 7.5 km/s while the Waystation Scenario is restricted to 8 km/s.

2.4 Asteroid Selection Process

Using an asteroid data file obtained from the Jet Propulsion Laboratory (JPL) and the above criteria, the selection of candidate asteroids proceeded as follows:

1. The initial selection required a maximum outbound ΔV of 8.0 km/s.
2. The number of asteroids selected in step 1 was reduced by studying the taxonomic classifications and choosing the desirable asteroids.
3. The final candidates were chosen based on the frequency of launch windows, the duration of stay times and the frequency of return windows.

For the first step, all 4800+ asteroids were analyzed with the FORTRAN program ASTOUT.FOR (see Appendix B). This program calculated the necessary ΔV to leave LEO and insert into the asteroid's orbit. It varied launch dates between 1/1/1997 and 12/31/2010 in 15 day increments and time of flights (TOF) between 150 and 450 days in 10 day increments to find the lowest ΔV combination. The program found eight asteroids that met the ΔV criteria (shown in Table 3).

Table 3: The Eight Asteroids Selected by ASTOUT.FOR.

Number	Name	a (AU)	e	i (deg)	Taxonomic Class	ΔV (km/s)
1943	Anteros	1.43	0.256	8.70	S	7.04
3505	Camel...	1.53	0.365	8.20	?	6.97
3637	1959LM	1.34	0.379	3.30	?	7.63
3650	1982MR	1.21	0.322	2.69	?	7.81
3677	1982DB	1.49	0.360	1.42	?	7.07
3679	1982HR	1.20	0.322	2.68	?	5.91
3684	1980YS	1.82	0.321	2.27	?	7.57
3687	1981CW	1.87	0.368	4.77	?	7.51

Unfortunately, most of the taxonomic classification of these asteroids are unknown. The Hubble Space Telescope could be used to obtain the necessary information.(1)

Next, launch opportunities, stay times, return opportunities and total ΔV for the mission were investigated for these eight asteroids.

2.5 Asteroid 3677

From the selection criteria, asteroid 3677 was selected for development of the exploration mission and spacecraft for reasons listed below:

- Accessible launch window with a reasonable stay time before the return flight.
- From the analysis of the ΔV budget for 3677, an explorer craft developed for 3677 could also perform missions to other candidate asteroids.

Figures 1 and 2 show one set of launch and return windows available for asteroid 3677.

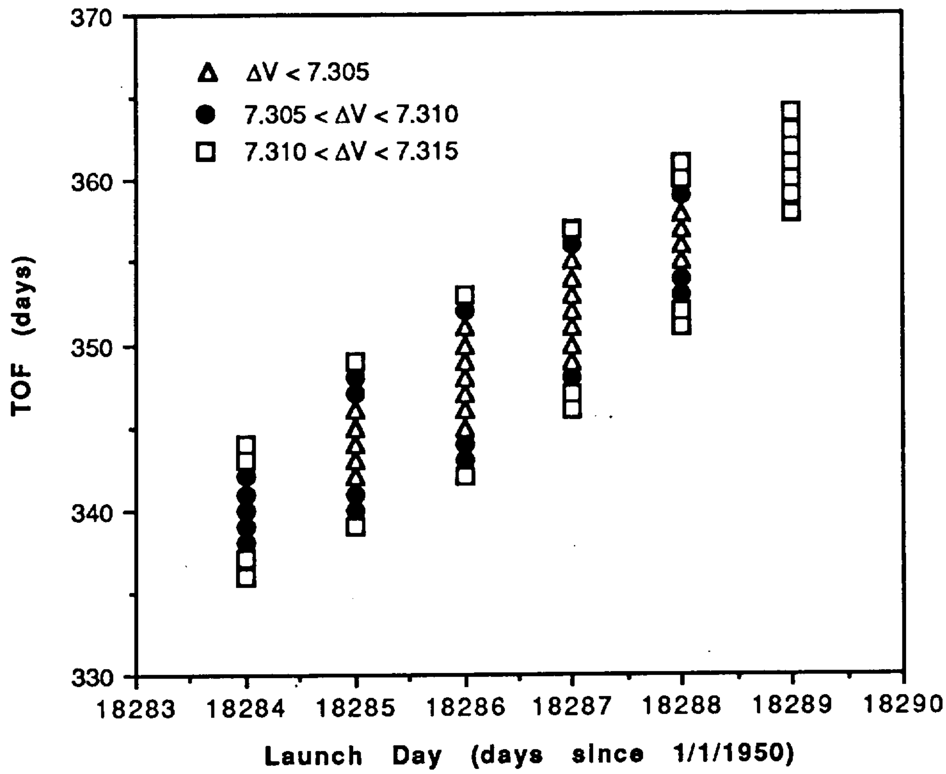


Figure 1. Launch Window for 3677.

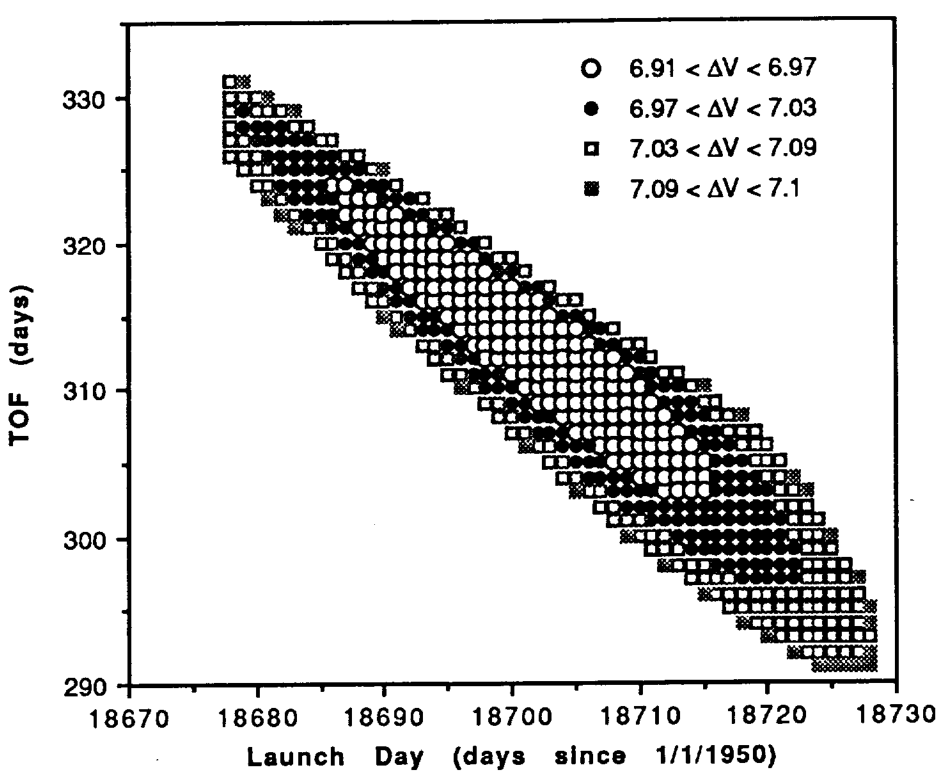


Figure 2. Return Window for 3677

3.0 Asteroid Exploration, the First Step

As stated before, the first step towards utilization of asteroids is exploration. This section outlines a plan for asteroid exploration that fulfills the requirements allowing utilization to proceed. The plan includes the mission objectives, assumptions and scenarios for asteroid exploration.

3.1 Mission Objectives

The overall objective for asteroid exploration is to determine the mineral composition of the different taxonomic classes so that utilization of asteroid resources can begin. This can only be accomplished by sending an explorer craft to at least one representative asteroid of each major taxonomic class and returning a sample to Earth for study.

Each of the explorer craft have common mission objectives[1, 2]:

1. Return a 3kg sample of asteroidal material to an Earth orbit. The sample would consist of a coring plus surface material.

2. Determine the following properties of the asteroid using remote sensing:

    - Bulk: Size, shape, volume, gravity field, spin rate

    - Surface: Geology, morphology, texture, near asteroid dust, solar wind interaction

    - Internal: Mass distribution, magnetic field

3. Fly-by other asteroids and perform intermediate studies en route to the selected asteroid to maximize scientific return.

3.2 Mission Assumptions

Some assumptions are necessary for an asteroid exploration/sample return mission:

- The material and miniaturization technology used in the development of SDI's "brilliant pebbles" is available for use on the explorer spacecraft. A "brilliant pebble" is a two-stage kinetic kill vehicle orbiting in LEO that intercepts and destroys missiles. Although most information is classified, it is known that a "brilliant pebble" uses advanced light-weight composites, miniature avionics and advanced propulsion systems.

- Mariner Mk II components are available for use on the explorer spacecraft. The Mariner Mk II is a modular spacecraft developed by JPL to reduce development time and cost of future spacecraft.

- The Hubble space telescope (HST) is repaired and available. The HST would be used to verify or obtain information about candidate asteroids.

- Launch capabilities available today will be available in the near future.

- Space Station Freedom will be available to initially analyze the sample returned to Earth orbit. This is to protect the sample from potentially damaging stresses during re-entry. Once initial studies are performed at Freedom, samples could be taken to the Earth via the shuttle.

- NASA continues to grant permission to launch vehicles with potentially hazardous materials aboard (ie RTG or toxic propellants).

3.3 Layout of the Hawking Explorer

The explorer spacecraft has been named for Dr. Stephen Hawking in honor of his efforts to expand the limits of man's understanding of the Universe.

The Hawking Explorer Spacecraft consists of three main sections as shown in Figure 3: the orbiter, the sample return craft (SRC) and the lander. The Hawking Explorer Spacecraft is broken up in this manner because each component vehicle fulfills a specific part of the overall mission objectives. Each section of the spacecraft is described briefly below and in detail in Sections 4 - 7.

The orbiter vehicle of the spacecraft is essentially a Mariner Mk II spacecraft stripped of its main propulsion system and tanks. Its part of the mission is to remotely observe the asteroid. The SRC is an enlarged "brilliant pebble". Its part of the mission is to return the sample safely back to Earth orbit. The lander is a new design. Its part of the mission is to collect samples for the SRC, provide a stable landing/coring platform and perform *in situ* studies of the asteroid.

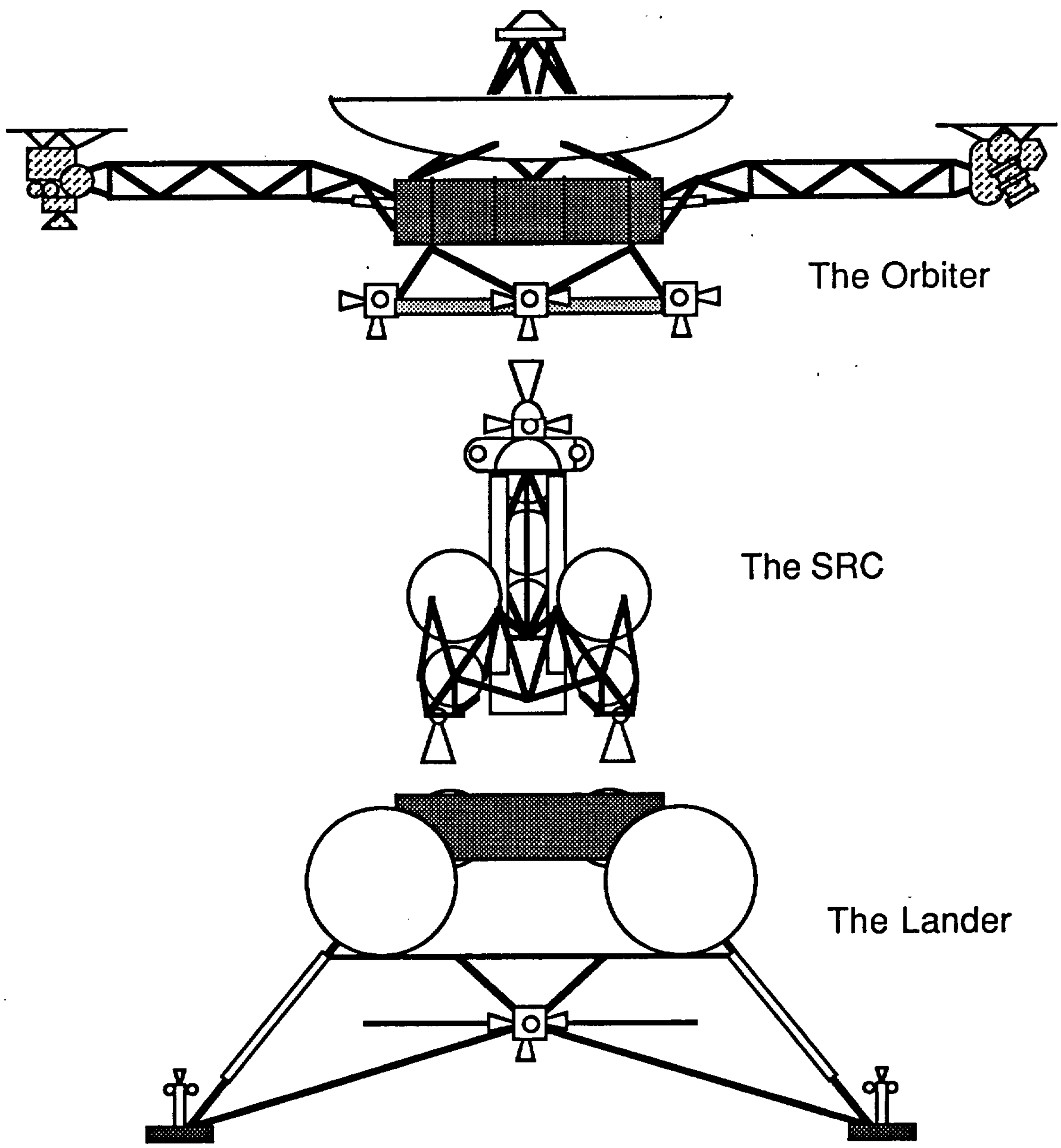


Figure 3: Exploded View of the Hawking Explorer.

3.4 Exploration Scenario

The main objective of asteroid exploration is to determine the mineralogical properties of asteroids through direct sampling. The Hawking explorer and its mission have been designed to meet these objectives. The Hawking mission scenario is broken into nine phases of operation:

Phase 1: The Hawking is launched into LEO orbit aboard an existing launch vehicle. Once in its circular parking orbit, the spacecraft and upper stage booster perform several systems checks in preparation for injection into the transfer trajectory. At the same time, ground-based tracking stations accurately determine the spacecraft's orbit. Figure 4 shows the Hawking spacecraft in launch configuration with upper stages.

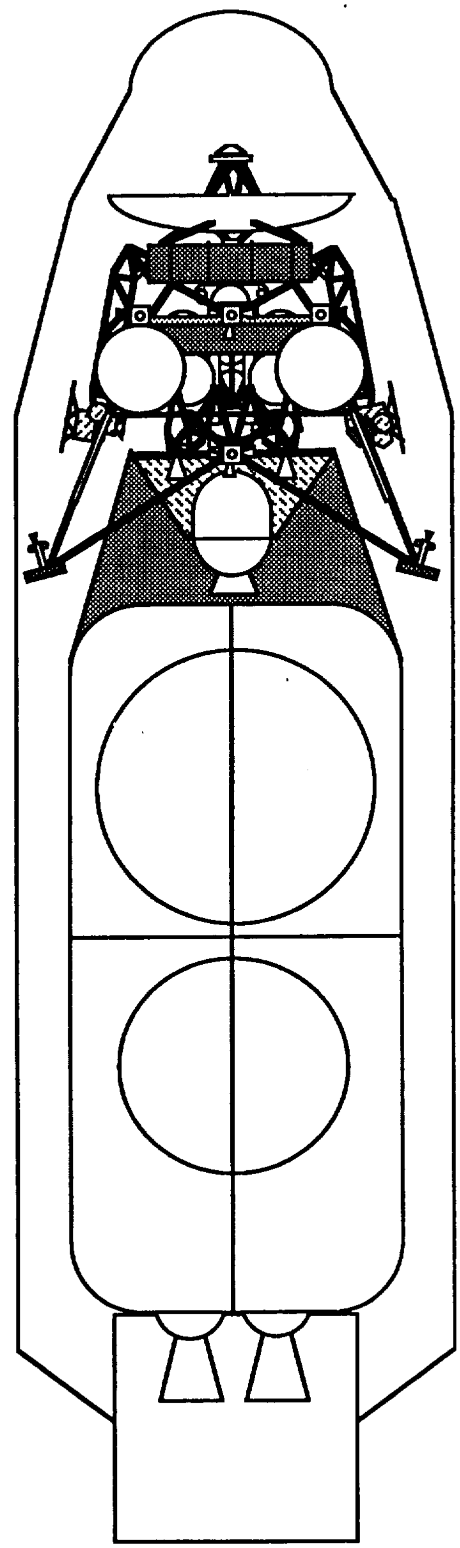


Figure 4: Hawking and Upper Stages Inside a Titan IV Shroud.

Phase 2: The Hawking is injected into the transfer trajectory by its upper stage. The upper stage is ejected upon burnout and the booms deployed. The spacecraft then activates its GNC systems, determines its attitude and points its high gain antenna at the Earth. Also, the spacecraft's subsystems are rechecked and calibrated.

Phase 3: During the interplanetary cruise, the spacecraft performs radio science experiments and studies of the solar wind. During the cruise and mapping phases of the mission, the spacecraft is controlled by the orbiter (see Figure 5).

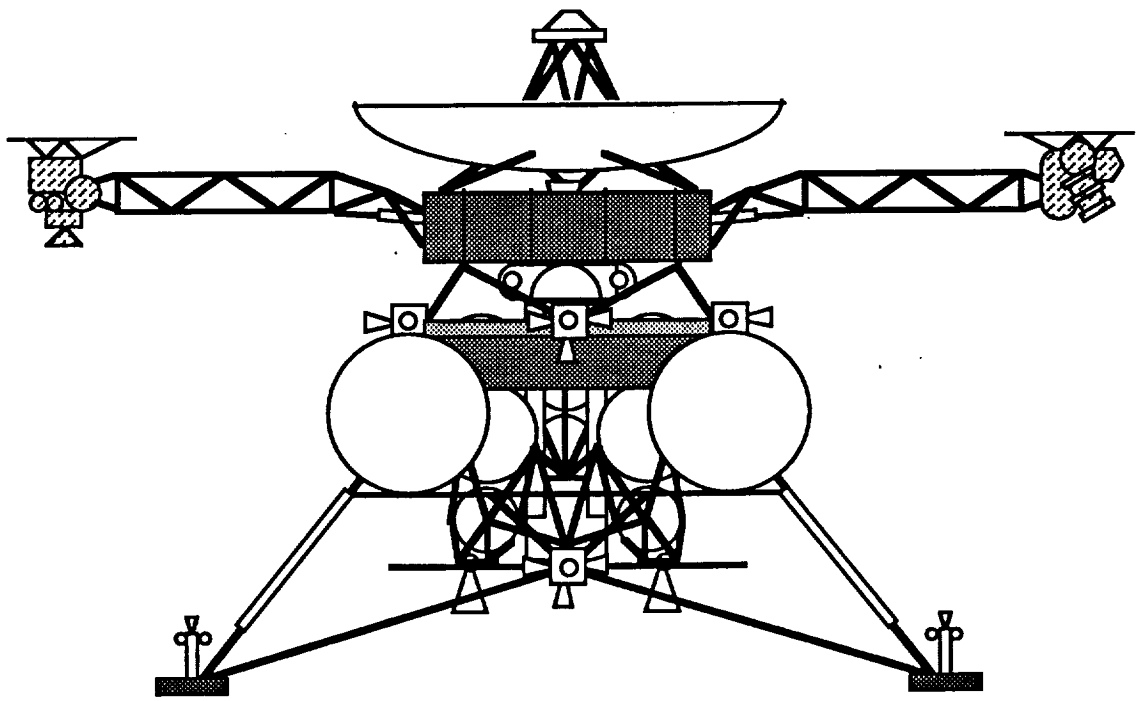


Figure 5: Hawking Explorer in Cruise Configuration (phase 3).

Phase 4: Hawking inserts itself into the asteroid's orbit, positioning itself several asteroid radii ahead of the body and slightly to the sun side. This position allows the spacecraft to map the asteroid, determine its rotational axis, and locate scientifically interesting features, all to help determine a desirable landing site. The mapping data is transmitted to science teams on Earth via ground receiving stations of the Deep Space Network (DSN). The coordinates of the final landing site chosen by engineers and scientists are transmitted to the spacecraft.

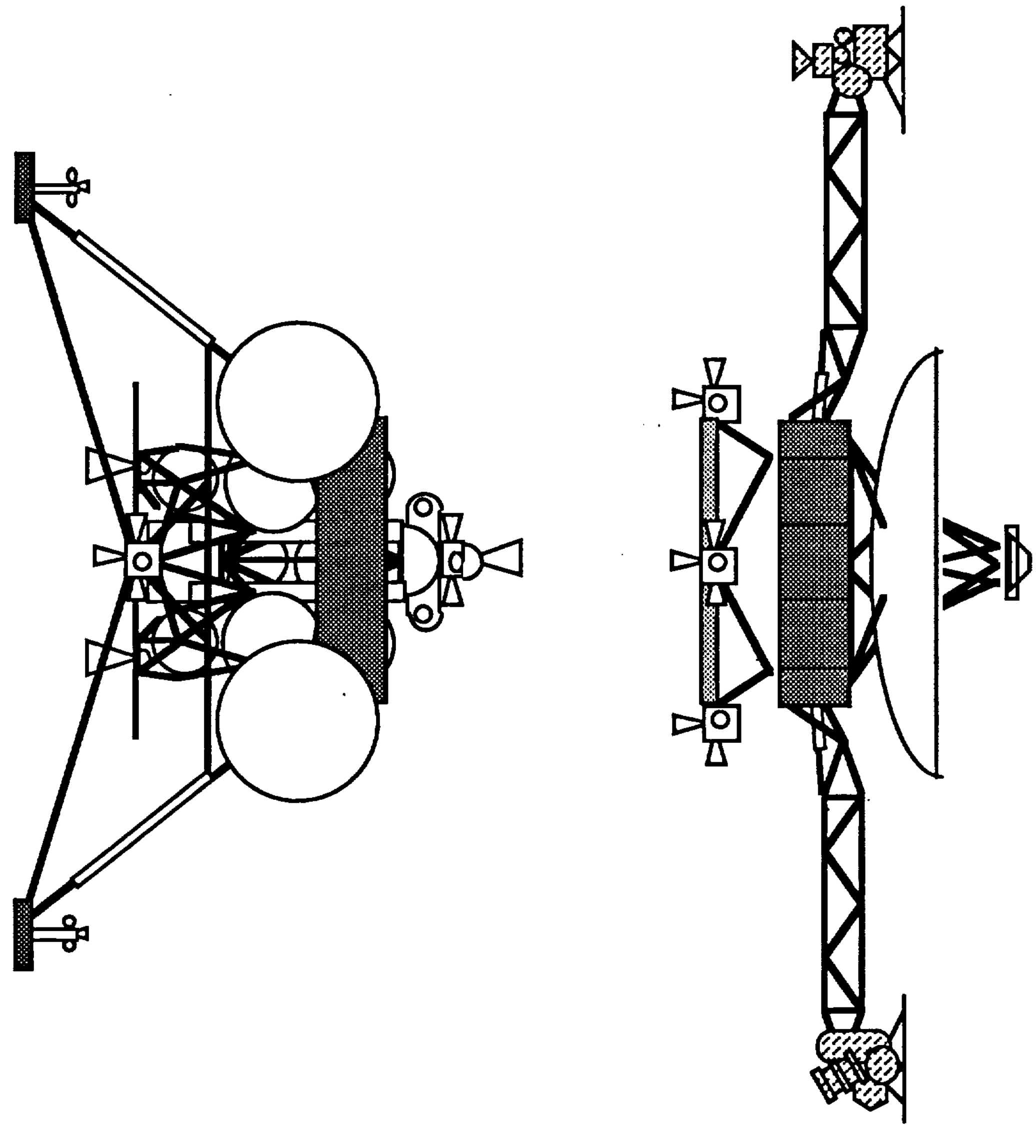


Figure 6: Orbiter and Lander/SRC Sections Separate (Phase 5).

Phase 5: Once a landing site is chosen, the lander/SRC separates from the orbiter (see Figure 6), which remains several asteroid radii away to serve as a relay station for the lander/SRC and to provide reconnaissance for the rovers. The lander/SRC approaches and docks with the asteroid using its attitude control thrusters. During the approach to the asteroid's surface, the lander/SRC is guided by the SRC's computers and the lander's attitude control thrusters. Because of the micro-gravitational field of the body, it is necessary to fasten the lander to the asteroid using a drilling mechanism located lander's feet.

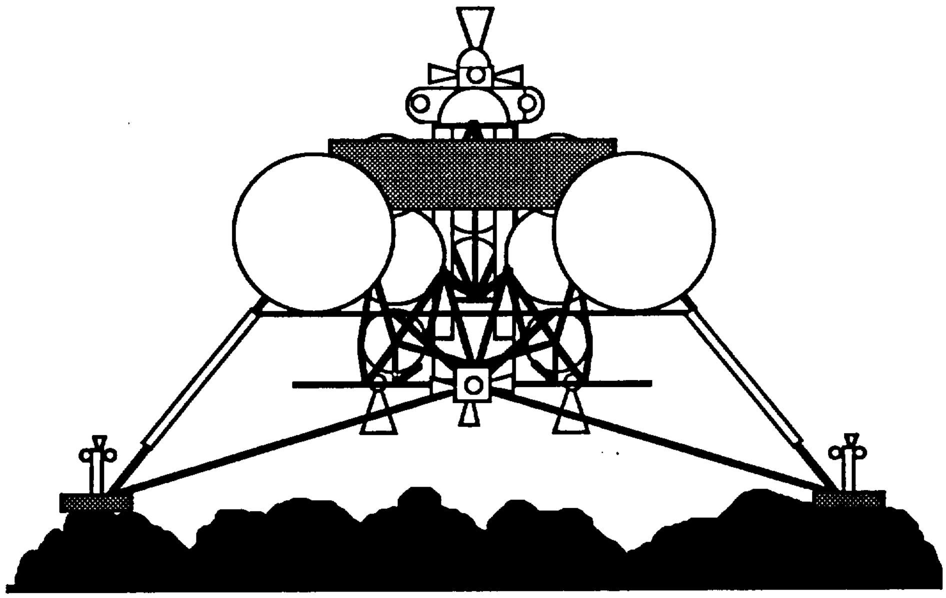


Figure 7: Lander/SRC Configuration on the Asteroid Surface (Phase 6).

Phase 6: The lander/SRC establishes radio communications with the orbiter over their low gain antennas, rechecks their systems and begins deployment of the robotic arm and the micro rovers for sample collection (see Figure 7). Samples of scientific interest would be identified and retrieved by either the robotic arm or the rovers. Some analysis would be performed on the lander and the data relayed back to the Earth via the orbiter's high gain antenna. Samples would placed in the SRC, which would be launched either when its capacity has been reached or the launch window for return to Earth closes.

Phase 7: The SRC is launched from the lander in a non-destructive manner (ie., springs, pneumatic pistons) so the lander can remain intact to perform more analysis of the asteroid (Figure 8). At an altitude of approximately 0.5 km above the lander, the SRC rotates and fires the booster's engines to inject itself into the transfer trajectory back to Earth. The SRC travels behind the asteroid to avoid fouling the antennas of the orbiter and the lander with its exhaust.

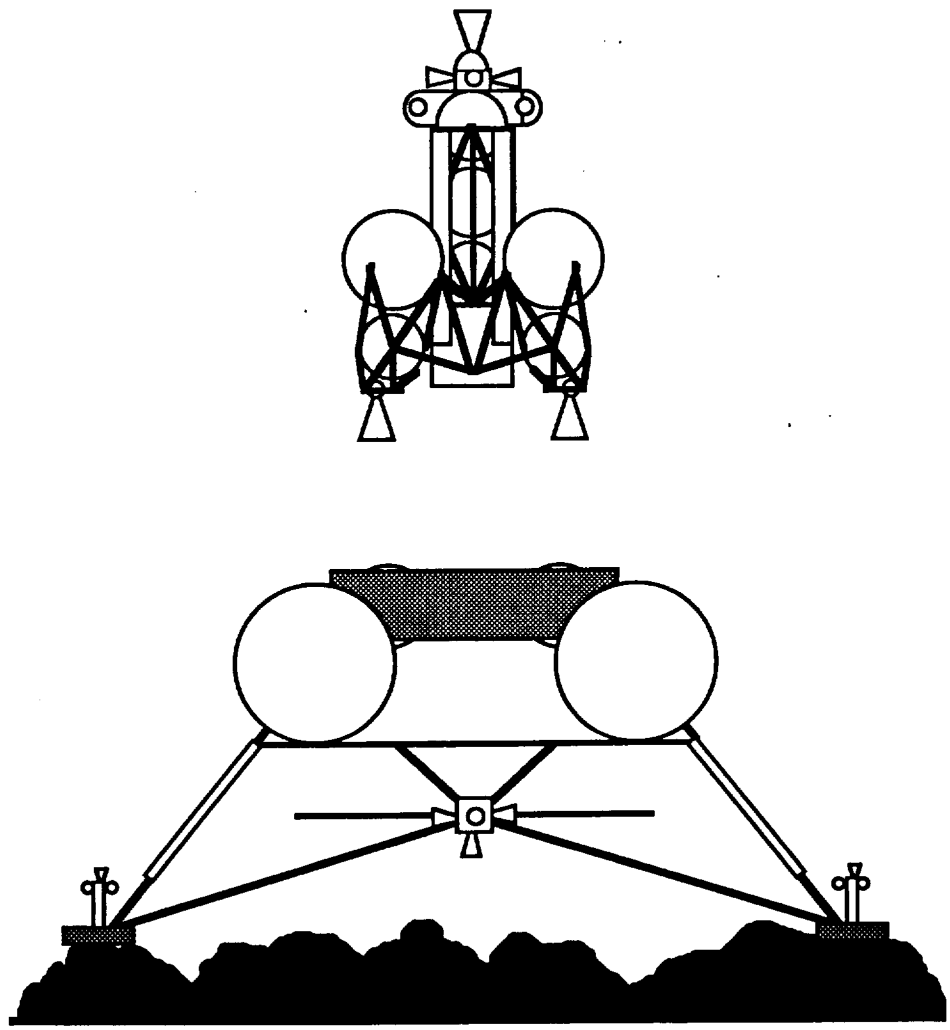
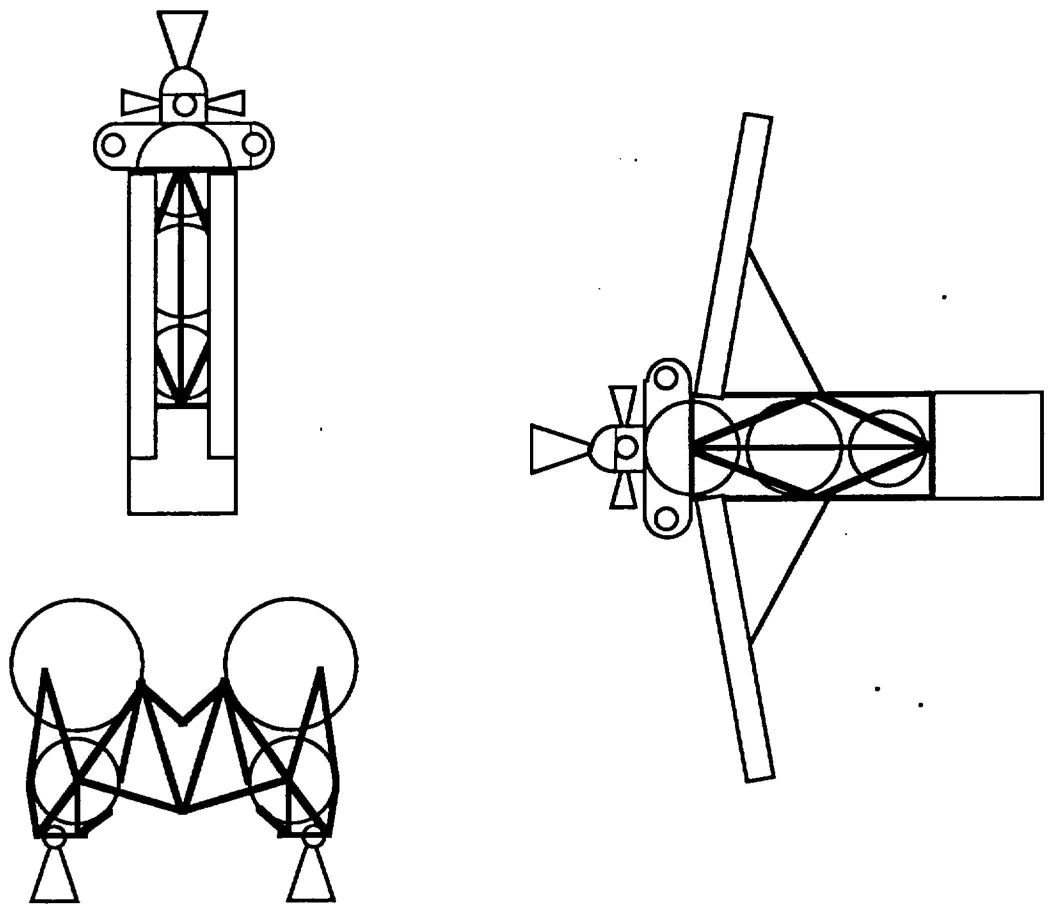


Figure 8: Non-Destructive Launch of the SRC from the Lander (Phase 7).

Phase 8: After the injection burn is completed, the booster stage is jettisoned and the communication antenna and solar panels are deployed (see Figure 9). During the interplanetary cruise of the SRC, the integrity of the sample is maintained by minimizing g-loads during maneuvers and keeping the sample at a low temperature.



SRC Boosters are Jettisoned   SRC Panels are Deployed

Figure 9: SRC Jettisons Boosters and Deploys Solar Panels (Phase 8).

Phase 9: Upon arrival at Earth, the SRC inserts itself into a highly elliptical orbit (approx. 300 km perigee altitude, eccentricity = 0.9). After accurate ground-based orbit determination, the SRC circularizes into LEO where it waits for pick-up by either the Space Shuttle or Space Station Freedom. To maintain the integrity of the samples, the g-loads during insertion burns are minimized by extending burn times.

Since the asteroid sample was formed in a micro-g environment and might have delicate crystal structures, it is important to maintain the integrity of the samples. Initial sample analysis on board Space Station Freedom is desirable because if the sample is brought down to Earth, the heat and forces associated with reentry could damage it. Once the initial studies of the sample are completed, some of the material would be brought to Earth (via the shuttle) for more in-depth analysis.

3.5 Mission Scenario to Asteroid 3677

Following the above scenario, a Hawking mission to asteroid 3677 was developed. For the flight to asteroid 3677, the Hawking spacecraft would be launched into a standard LEO orbit. On January 21, 2000 the Hawking would perform a 5.6km/s burn (via upper stages) that sends it on a transfer trajectory to the asteroid. During the 340 day cruise to the asteroid, the Hawking would perform standard radio science experiments.

The transfer trajectory of the Hawking is targeted to a point 50 km (100 asteroid radii) ahead of the asteroid's orbit. On 12/27/2000, the Hawking performs a 1.7 km/s burn to insert itself into the desired orbit. After the orbiter has completed an initial study of the asteroid and a landing site has been chosen (estimated 30 days), the lander/SRC section of the spacecraft separates from the orbiter and prepares to dock with the asteroid. Docking maneuvers are estimated to take 5 hours with a total ΔV of 0.2 km/s.

After 104 days from initial insertion, the SRC positions itself behind the asteroid in preparation for Earth return. On 4/10/2001, the SRC performs a 5 km/s burn that injects it into a transfer trajectory back to Earth. After 295 days, the SRC performs a 1 km/s burn at closest approach to Earth (300 km altitude) to insert itself into an orbit with an eccentricity of 0.9. A final burn of 3 km/s circularizes the orbit (300 km altitude), from which the sample is eventually retrieved. Table 4 summarizes the dates and activities.

Table 4: Information on the Mission to Asteroid 3677.

Phase	Activity	Activity Date	Duration (days)	ΔV (km/s)
1	Departure from LEO (Stage 1)	1/21/2000		4.0
2	Departure from LEO (Stage 2)	1/21/2000		1.6
3	Cruise to Asteroid & TCM		340	.5
4	Arrival and Insertion	12/27/2000		1.7
5	Study/Docked with Asteroid		104	.2
7	Departure from the Asteroid	4/10/2001		4.5
8	Cruise to Earth & TCM		295	.5
9	Arrival and Insertion	1/30/2002		1.0
9	Circularization to LEO			3.0

The launch date is chosen early in the window (see Figure 1) to maximize stay time on the asteroid. This allows for launch delays at the Earth, adequate time for the orbiter to study the asteroid (30 days), and ample stay time on the asteroid to gather samples.

Figures 10 illustrates the outbound trajectory to asteroid 3677 1982DB while Figure 11 shows the return trajectory.

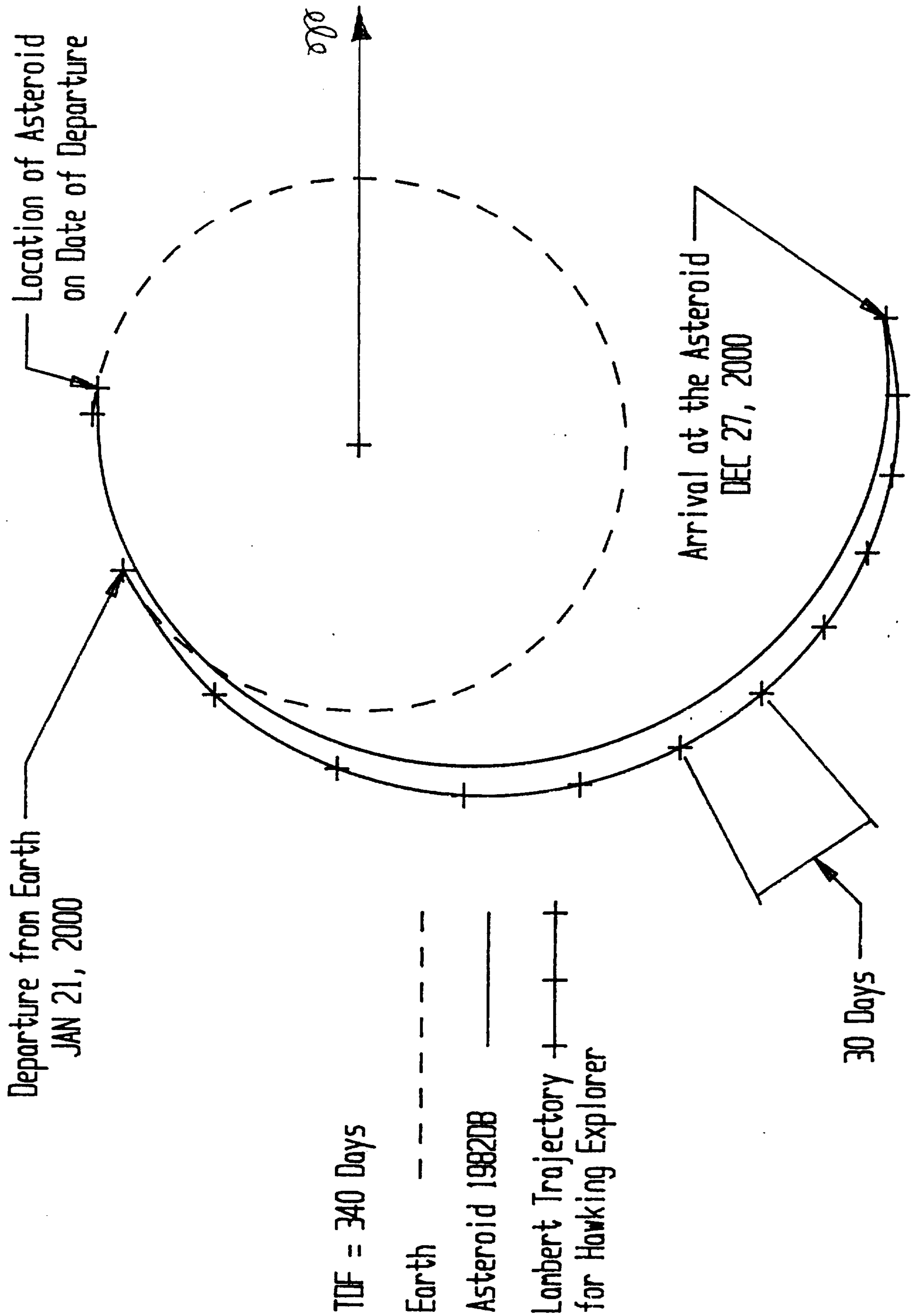


Figure 10: Outbound Trajectory to Asteroid 3677 1982DB.

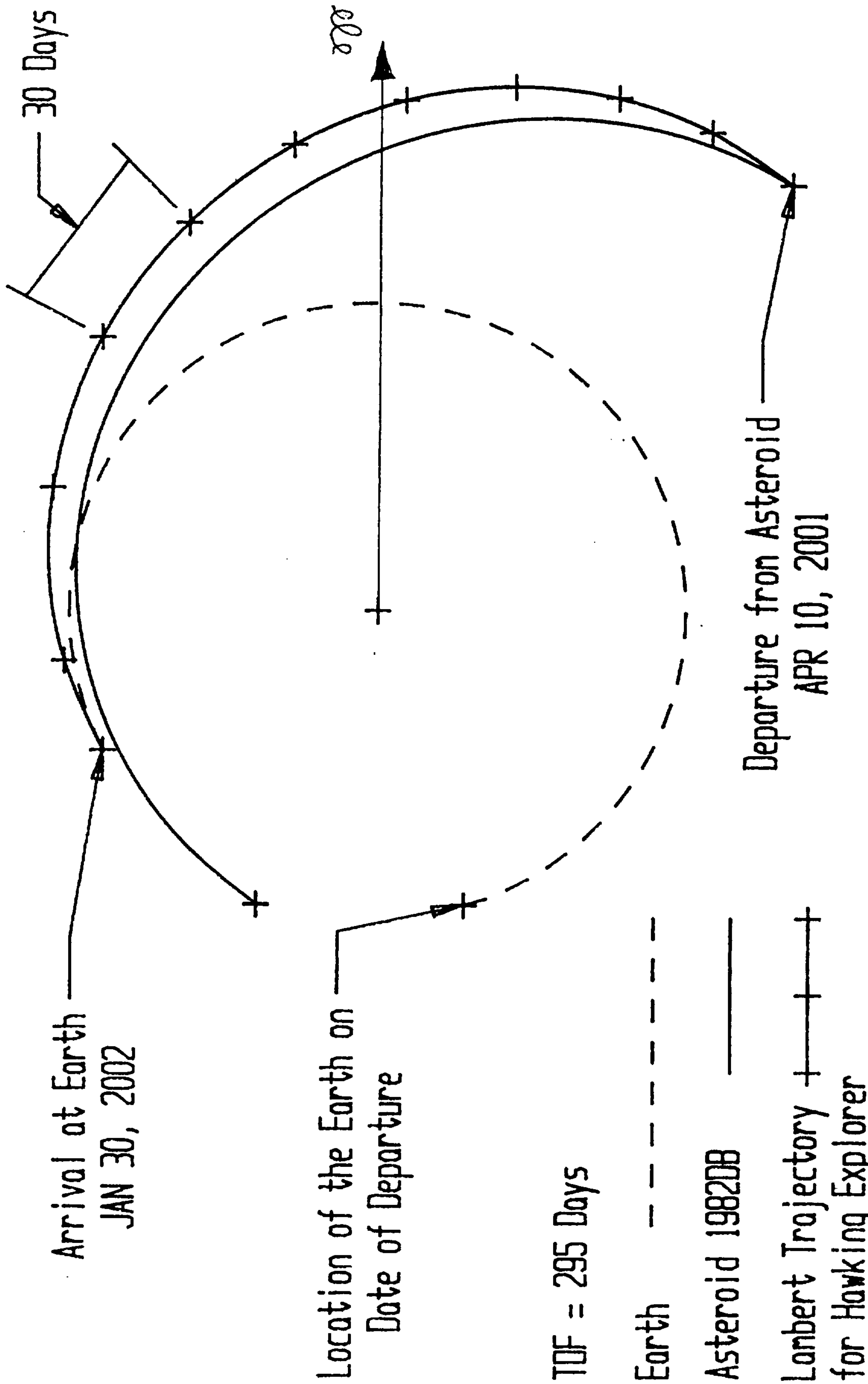


Figure 11: Return Trajectory from Asteroid 3677 1982DB.

4.0 The Hawking Explorer Design Philosophy

Since the main mission of the Hawking is to return an asteroidal sample to the Earth via the SRC, the spacecraft must carry fuel and structure that is not used until the end of the mission. Thus, Hawking's initial mass is higher than a normal explorer probe. The initial mass of the spacecraft is reduced by:

- Using a high-performance propellant
- Staging the spacecraft
- Integrating some subsystems

A chemically efficient propellant provides high combustion energy per kilogram burned (high I_{sp}), reducing the amount of propellant necessary for the total ΔV (thus reducing the total weight of the spacecraft). Staging throughout the mission, from Earth launch to asteroid departure, allows structure that is no longer useful (i.e. propellant tanks, casings, and engines) to be jettisoned from the spacecraft. Integration of subsystems allow for weight to be reduced because their is less redundancy.

Following the staging philosophy, the Hawking Explorer spacecraft is made up of three main vehicles:

- The orbiter
- The lander
- The sample return craft (SRC) and booster

This method increases the efficiency of each component vehicle because the weight for each phase of the mission is minimized.

Two subsystems that have been integrated between the three vehicles:

- The guidance, navigation and control system
- The propulsion system

4.1 Control System Integration

The responsibilities of guidance, navigation and control of the spacecraft during various phases of the mission are shared by the flight computers of the orbiter and the SRC.

During the initial cruise phase from Earth, the orbiter's flight computers control the spacecraft via the lander's attitude control thrusters and SRC's booster engines for trajectory correction maneuvers and the asteroid insertion burn.

Upon separation of the lander/SRC and the orbiter at the asteroid, the orbiter controls itself with its own attitude control thrusters. The SRC flight computers now command the lander/SRC during docking burns using the lander's attitude control thrusters.

During asteroid departure, the SRC commands its own attitude control thrusters and the booster's main engines. After the booster is jettisoned, the SRC uses its attitude control thrusters and main engine to perform TCM's and complete the final two Earth insertion burns.

4.2 Propulsion System Integration

The requirements of the propulsion system during the mission include orientation and docking maneuvers, trajectory correction maneuvers (TCM's), orbital transfer burns and planetary insertions. These maneuvers require a total ΔV of 17 km/s.

The spacecraft uses chemical propulsion systems and must carry the fully-fueled SRC as payload until the last two burns of the mission. Minimizing the fuel mass of the SRC and its preceding stages reduces Hawking's total initial mass. High I_{sp} propellants can help reduce the initial spacecraft mass, increasing the chances for the mission to use an existing launch vehicle.

4.2.1 Supporting Assumptions

Although stated in the mission assumptions section, these assumptions are re-emphasized because they are crucial to the mission:

- Highly toxic and volatile propellants are permitted to be launched to LEO with existing launch vehicles.
- Advanced Liquid Axial Stage (ALAS) technology developed for SDI's "brilliant pebbles" can be used.(1)
- Corrosion-resistant materials exist or can be developed to sustain propellant tank life in the corrosive environment created by the fuel and oxidizer.

Most propellants that produce I_{sp}'s greater than 350 seconds are very toxic. If spread into the atmosphere due to improper handling or launch system failure, they are lethal to human beings. There are legal questions as well as public opinion to consider when launching hazardous materials. This report assumes that restrictions on the use of toxic propellants are not insurmountable.

Presently, SDI's Brilliant Pebbles program is supported by Aerojet's ALAS technology. ALAS uses lightweight carbon overwrap propellant tanks as well as lightweight high performance thrust nozzles and combustion chambers. Regulators and valves have been miniaturized for the ALAS to further reduce its mass. Also, the ALAS has a composite support structure to maximize strength-to-weight characteristics. This technology is currently being validated for full scale development of Brilliant Pebbles as well as for future prospects of the Space Exploration Initiative (SEI).(1)

Lightweight and durable propellant tanks are necessary for any space exploration mission. Oxidizers and fuels can be very corrosive to low density metals and organic materials that are currently used to make propellant tanks. Many of the metals with corrosion-resistant characteristics are very dense, and to minimize weight it is only practical to use them as cladding (or lining) for propellant tanks. However, plastics and polymers, used as coatings or thin films, have also proved to be an effective means of corrosion prevention.

4.2.2 Hawking Propulsion Scenario

Table 5 outlines the 9 propulsion phases of the mission to asteroid 3677. Note the initial mass of the Hawking (on the launch pad) including upper stages is 21,537 kg. Also, the total ΔV of the mission is 17 km/s and the mass at arrival at the Earth is 61 kg.

Table 5: Propulsion Requirements for a Hawking Mission to Asteroid 3677.

Phase	Activity	Activity Date	ΔV	I_{sp}	Fuel Mass	Hawking Mass
		mm/dd/yy	km/s	s	kg	kg
1	Departure	1/21/2000	4.0	446	12,904	21,537
2	Departure		1.6	292	2,643	6,175
3	Cruise		.5	445	361	3,333
4	Arrival	12/27/2000	1.7	445	959	2,972
5	Docking		.2	445	50	1,111
7	Departure	4/10/2001	4.5	445	411	639
8	Cruise		.5	445	19	172
9	Arrival	1/30/2002	1.0	445	31	153
9	Circularize		3.0	445	61	122

During the initial cruise phase from Earth, the orbiter's flight computers control the spacecraft via the lander's attitude control thrusters and SRC's booster engines. The lander's 5-10 N attitude control thrusters and SRC booster's two main engines (1,250 N) use propellant from the tanks on the lander. Figure 12 shows the configuration of the propulsion system for Phase 3 of the mission. Solid lines indicate active propulsion section while dashed lines inactive.

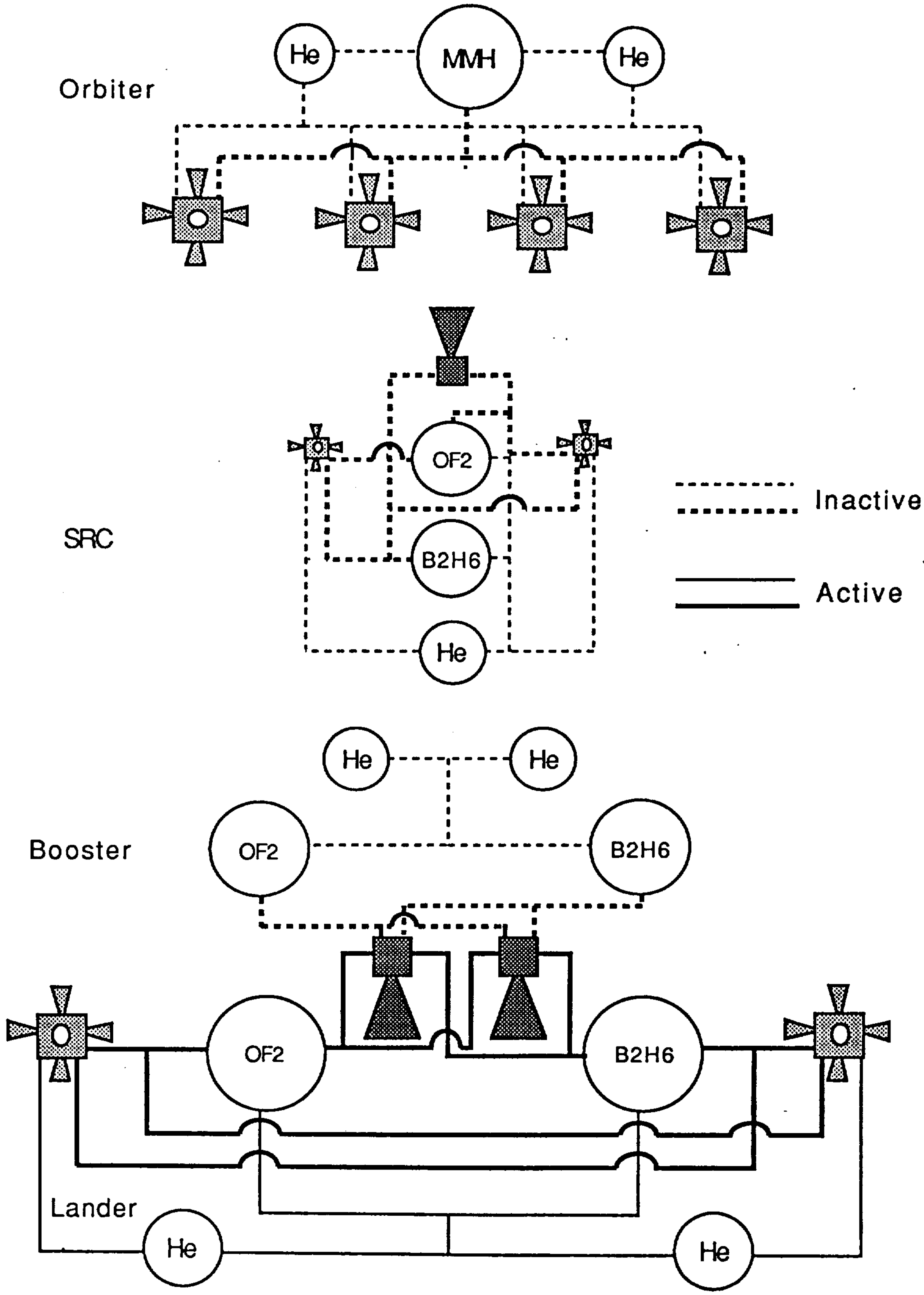


Figure 12. Schematic of Hawking Propulsion System for Phase 3 & 4.

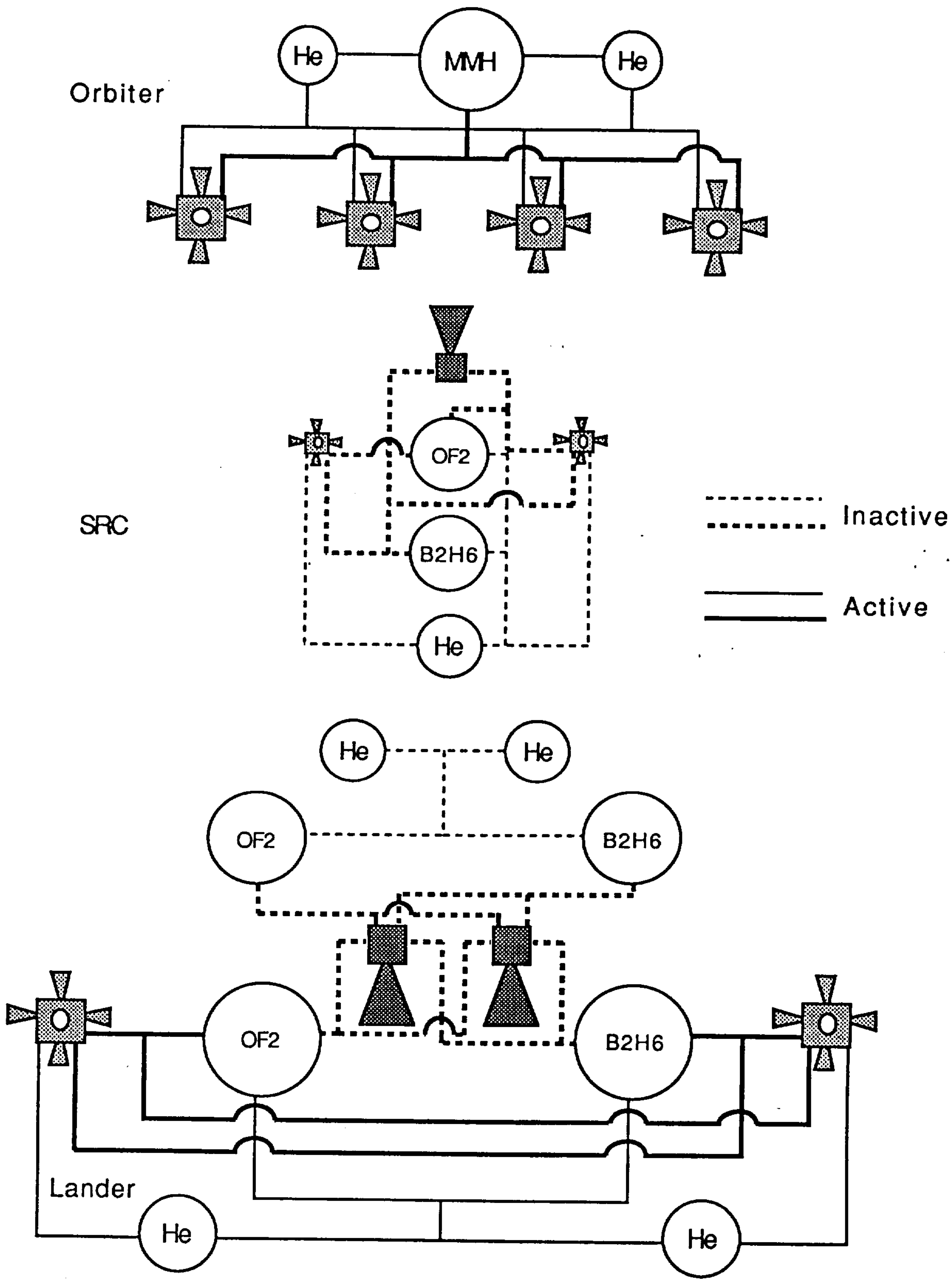


Figure 13. Schematic of the Orbiter and lander/SRC Propulsion Systems.

Upon separation of the lander/SRC from the orbiter, the orbiter controls itself with its own attitude control thrusters. These consist of 5-10 N monopropellant hydrazine thrusters and 1-2 N cold gas helium jets. The orbiter carries an estimated 30-40 kg of hydrazine onboard. Figure 13 shows the orbiter and the lander/SRC propulsion systems separate and active during Phase 5.

The SRC flight computers now control the lander/SRC during docking maneuvers using the lander's attitude control thrusters which draw their fuel from the lander's tanks (see Figure 13). When preparing to leave the lander, the SRC disconnects the booster's engines from the lander's tanks and switches them to the booster's tanks. The lander's tanks are left with the lander on the asteroid surface.

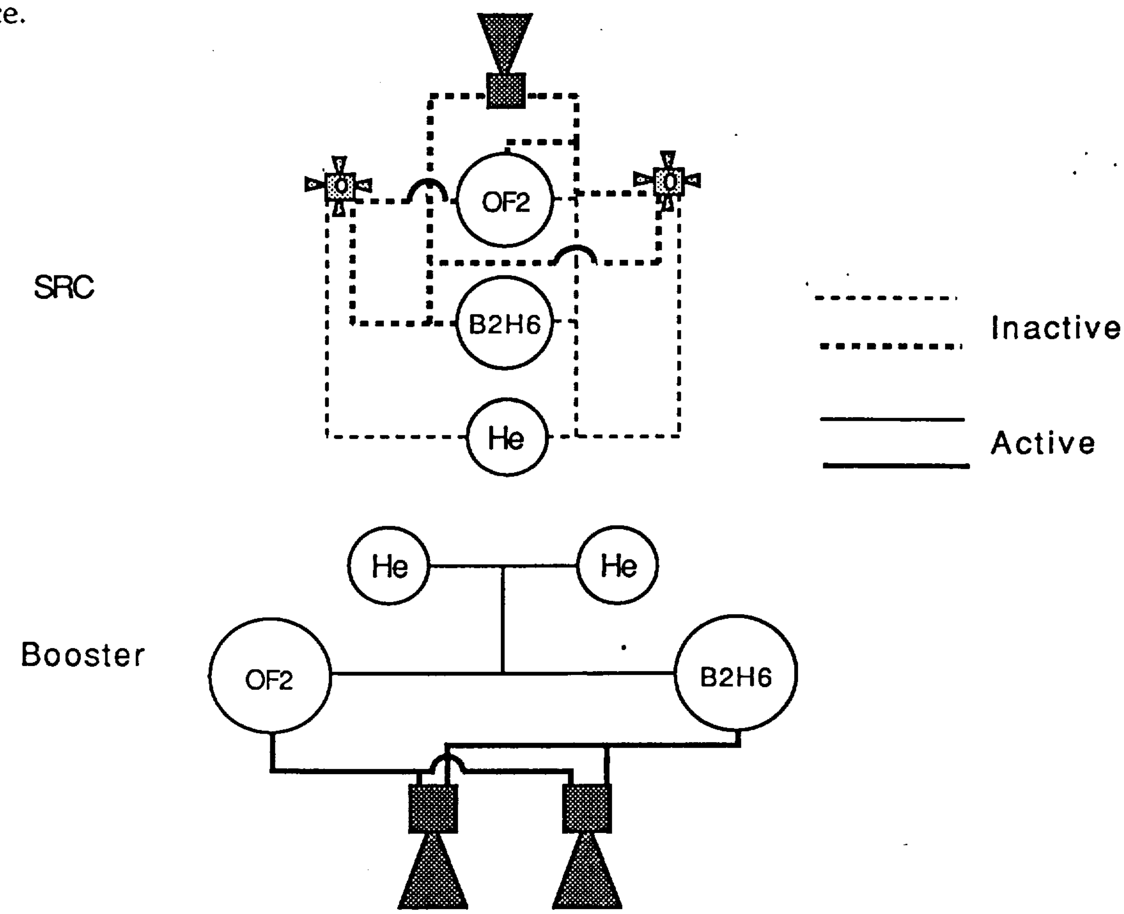


Figure 14: Schematic of the Booster and SRC Propulsion System during Phase 7.

During asteroid departure, the SRC commands its own attitude control thrusters and the booster's main engines as shown in Figure 14. The SRC can use its 1-2 N cold gas helium jets or 5-10 N OF_2/B_2H_6 thrusters to orient itself before and after the main engines perform the Earth transfer burn.

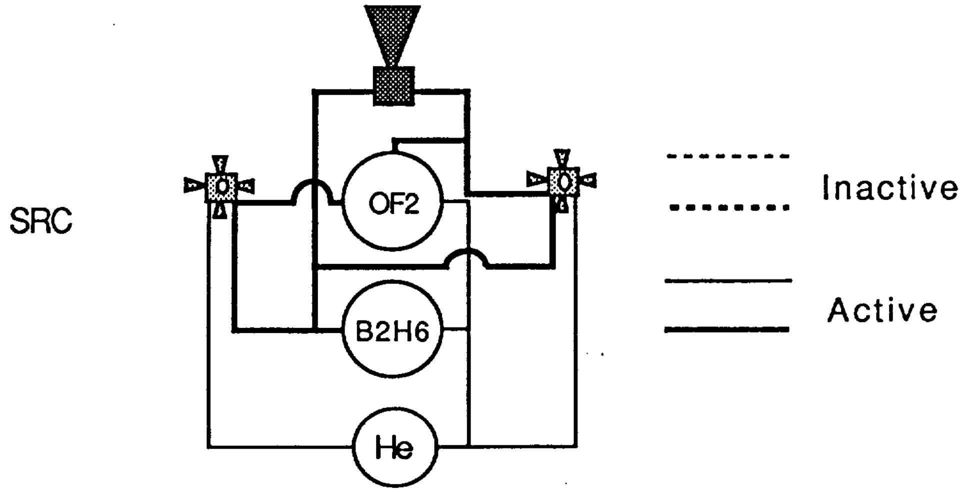


Figure 15: Schematic of the SRC Propulsion System during Phases 8 & 9.

After the booster separates, the SRC commands its attitude control thrusters and 500 N main engine to perform TCM's and the final Earth insertion burns (see Figure 15). The fuel for these maneuvers comes from the SRC's onboard propellant and pressurant tanks.

4.2.3 Propellants

Hawking's main propulsion system uses oxygen difluoride (OF_2) liquid oxidizer and diborane (B_2H_6) liquid fuel. At an oxidizer/fuel ratio of 3.98 and a rated combustion chamber pressure of 1,000 psia, this propellant combination produces an I_{sp} of 445 seconds.(2)

Oxygen difluoride is the second most powerful propellant oxidizer available (next to fluorine). Table 6 presents some physical characteristics of OF_2 and B_2H_6. Both propellants are space storable, and can be stored on Earth at the same temperature (~-240 °F/-151 °C). Since both propellants are insensitive to shock they should withstand any harsh conditions or hostile environments experienced during the mission.(3)

Table 6. Propellant Characteristics[3]

Propellant	Mol. Wt	Density[a] (g/cm^3)	Freezing Point[b] (°F)	Boiling Point[b] (°F)	Shock Sensitivity
Oxygen Difluoride	54	1.52	-371	-170	Insensitive
Diborane	28	0.430	-266	-55	Insensitive

[a]At normal B.P. or 71°F, whichever is less. [b] @ 100 psia.

There are two disadvantages in using OF_2 and B_2H_6:

- Both are very toxic substances requiring strict handling, storage, and management techniques.
- Both are very corrosive -- especially oxygen difluoride -- to many ferrous metals.

Both propellants are pressurized with helium, an inert low molecular weight gas that is commonly used as pressurant on space missions and as propellant for cold gas attitude control jets.

4.2.4 Propellant Tanks

Since Hawking requires only moderate thrust levels (<50,000 N) to complete its mission, a blowdown system is used instead of a regulated system. The additional mass required by a regulated system does not save weight overall unless thrust and total impulse requirements are high.[4]

Hawking's total impulse is 8,254,878 N-s, corresponding to a total fuel mass of 1,892 kg. A high total impulse can increase a blowdown propulsion system's mass relative to a regulated system, due to the large mass of fuel that needs to be pressurized.

All propellant tanks for the Hawking spacecraft are spherical. Spherical tanks are more efficient under high pressures because they require less surface area and wall thickness than cylinders. The combustion chamber pressure required for 445 seconds I_{sp} and 1,250 N thrust is 1,000 psia. Assuming a 50% pressure drop across propellant lines and injector plates and an 8:1 blowdown pressure ratio, the pressure of Hawking's propellant and pressurant tanks are estimated at 2,000 psia and 16,000 psia, respectively. These unusually high pressures for the blowdown system could be detrimental to minimizing the overall mass of the spacecraft, even with composite overwrapped tanks.

4.2.5 Material Requirements

Hawking uses corrosive propellants at unusually high pressures, so titanium propellant tanks which are standard on the Mariner Mark II spacecraft cannot be used. This inapplicability is shown by large titanium tank masses calculated in the NROCKET2.FOR program, discussed in Appendix C.

Experiments performed on high pressure compound composite/steel cylinders, at the U.S. Army Armament Research, Development, and Engineering Center, have produced overall weight savings of 34% -- as opposed to all-steel cylinders -- with 68% of the wall thickness being composite. Thus, carbon fiber overwrap technology, similar to that used for ALAS propellant tanks, should be seriously considered as an alternative to titanium tanks.(5)

The design of Hawking's propellant tanks and engines is based on the following material criteria:

- Low density, high yield strength
- High resistance to corrosion
- Good thermal characteristics (i.e. creep resistance, thermal reflectivity)

Table 7 presents some physical properties of materials being considered for use in Hawking's propulsion system components. Each of these materials has a specific function in the operational life of the component.

Table 7. Some Materials and their Properties(6),(7),(8)

Material	Density (kg/m^3)	Tensile Strength (MPa)	Coef. Thermal Expansion ($10^{-6}/°C$)	Corrosion Resistance to Oxidizers	Use
Titanium Alloys	4,500	760-900	8-10	Fair	Secondary pressure walls of propellant tanks
Aluminum Alloys	2,600-2,800	400-500	13-14.2	Fair	Secondary pressure walls of propellant tanks
Inconel	8,250	1,200	13	Good	Combustion chamber walls, tank inner-wall cladding for corrosion resistance
Teflon	2,150	27.6 max	----	Outstanding	Coatings, linings, or components to resist corrosion
Carbon Fiber Epoxy	1,550	579	-----	Unknown	Provide high strength and low weight for propellant tanks

4.2.6 Hawking Mass Breakdowns

The component vehicle masses of Hawking are calculated using the NROCKET2.FOR FORTRAN code. A complete explanation of the code and its results is located in Appendix C. Table 8 presents propellant and tank/engine masses of the component vehicles. These masses are based on propellant mass fractions assumed for the type of system (liquid or solid).

Since ALAS technology is applied to the design of the two 1,250 N main engines as well as the SRC booster and lander propellant tanks, mass fractions of 0.85 or greater for the booster and lander are assumed. The SRC's excellent mass fraction (>.90) results from using Brilliant Pebbles/ALAS miniaturization and lightweight materials technology. The mass fractions of Hawking's upper stages 1 (the Centaur-G) and 2 (Star 60 series solid motor) are known.

Table 8. Masses of Hawking's Component Vehicles

Component Vehicle	Propulsion System Type	Mass Fraction	Propellant Mass (kg)	Propulsion System Mass (kg)	Component Vehicle Mass (kg)	Total Mass (kg)
SRC	Liquid	.93	111	8	61	172
SRC Booster	Liquid	.88	411	56	56	467
Lander	Liquid	.88	1,370	186	238.7	1,434
Upper Stage 2 (Star 30)	Solid	.93	2,643	199	199	2,841
Upper Stage 1 (Centaur-G)	Liquid	.84	12,904	2458	2458	15,362

4.4 Launch Systems

The selection of the launch system for the mission was based on the following criteria:

- Compatibility with the upper stage booster
- Capability to boost mass of spacecraft and upper stage to LEO
- Sufficiently large shroud dimensions to accommodate spacecraft and upper stage booster.

4.4.1 Upper Stage Selection

It was necessary to select the upper stage boosters first, so that the total boosted mass could be used in the selection of the launch vehicle. In choosing each of the upper stage vehicles, the primary criterion was a high specific impulse to reduce fuel mass. Table 9 shows three boosters which were considered for the first stage and two boosters for the second stage. The Centaur-G was chosen because of its specific impulse of 446 sec. Although both second stage boosters are suitable, the Orbus Series 6 was chosen because of its higher I_{sp}.

Table 9: Upper Stage Boosters (9)(10)(11)

Upper Stage 1	I_{sp} (sec)	Burn Time (sec)	Thrust (N)	Mass (wet) (kg)
Centaur-G	446	609	147,000	19,200
IUS: SRM-1	292.9	153	200,000	14,800
SRM-2	300	104.8	81,200	
TOS	294	150	200,000	10,800
Upper Stage 2				
Star 63D	283	118	118,811	3,507
Orbus 6,6E,6S	302	101	110,072	2996

4.4.2 Launch Vehicle Selection

Combining the masses of the Centaur-G (15,362 kg), a solid motor (2,842 kg) and the Hawking spacecraft (3,333 kg), resulted in a total launch pad mass of 21,537 kg. The payload adaptor mass was estimated at 450 kg. Thus, the primary selection criteria for the launch vehicle were compatibility with the Centaur-G and a 21,987 kg boosted mass capability. Table 10 shows three systems and their boost capability.

Table 10: Possible Launch Systems.(9)(12)

Launcher	Developer	Boosted Mass to LEO (kg)
Titan IV SRMU	Martin Marietta	22750
Energiya	USSR	100000
STS	Rockwell International	22650

Although the Energiya has the greatest payload capability, obtaining necessary export licenses and congressional approval could delay the launches. These difficulties lead to a decision to use only American launch services.

The STS had to be eliminated because the danger of launching with RTG's and toxic fuels is unacceptable for a manned vehicle. Thus, the Titan IV-SRMU was selected as the only appropriate launch vehicle.[12]

Figure 4 shows the configuration of the Hawking spacecraft in the payload fairing of the Titan IV launch vehicle. The booms are shown folded so that the spacecraft will fit inside the shroud.

5.0 The Orbiter

As stated before, the orbiter is essentially a Mariner Mk II spacecraft without the major propulsion subsystem. The Mariner Mk II series was chosen because the high degree of modularity allows easy adaptation of the basic spacecraft to a variety of missions. Also, since there will be 5 - 10 exploration missions, adapting Mariner Mk II components to the Hawking spacecraft will help lower the overall cost of the missions.

5.1 Orbiter Mission Objectives

The orbiter has several mission objectives:

- Control the spacecraft during Earth departure, cruise and asteroid insertion phases (1 through 3).
- Perform radio science experiments en route to the asteroid.
- Perform an initial survey of the asteroid to help determine a landing site.
- Act as a communications relay between the lander and the Earth.
- Stay with the asteroid to continue studies of the asteroid, space, sun, etc. after the SRC has departed.

5.2 Scientific Instruments

In order to meet the scientific requirements of its mission, the orbiter must carry several instruments. The instruments and their functions are listed below in Table 11. They were chosen based on similar asteroid exploration missions.[1]

Table 11: Orbiter Instruments.[1]

Function	Instruments
Ambient background/interaction	DC magnetometer and gradiometer
	AC magnetometer
	Plasma wave detector
	Dust detector
	Cosmic-ray telescope
	Gravity gradiometer
Asteroid observation	Imaging telescope (TV)
	IR, UV and visible spectrophotometers
	Photopolarimeter
Gas envelope detection	Low energy plasma analyzer
	Ion mass spectrometer

5.3 Power

The orbiter is powered by two RTG's that each provide 284 W of power and weigh 56 kg. The RTG's were chosen primarily because of their long life, proven reliability in space and use the Mariner Mk II is designed to use this power source.[2]

5.4 Communications

The orbiter communications system consists of a high gain antenna (HGA) and a low gain relay antenna. These antennas are used to perform the orbiter's communications functions throughout the mission.

5.4.1 Communication Through the Phases

During Earth departure, cruise, and asteroid insertion portions of the mission, the orbiter's HGA receives all command and control data and transmits all telemetry and status data to Earth. After the lander/SRC and the orbiter separate and the lander/SRC docks with the asteroid, the orbiter communication system has two functions. First, the HGA continues to receive and transmit scientific data gathered by the orbiter. Second, it serves as a communications relay between the Earth and the lander/SRC (see Figure 16).

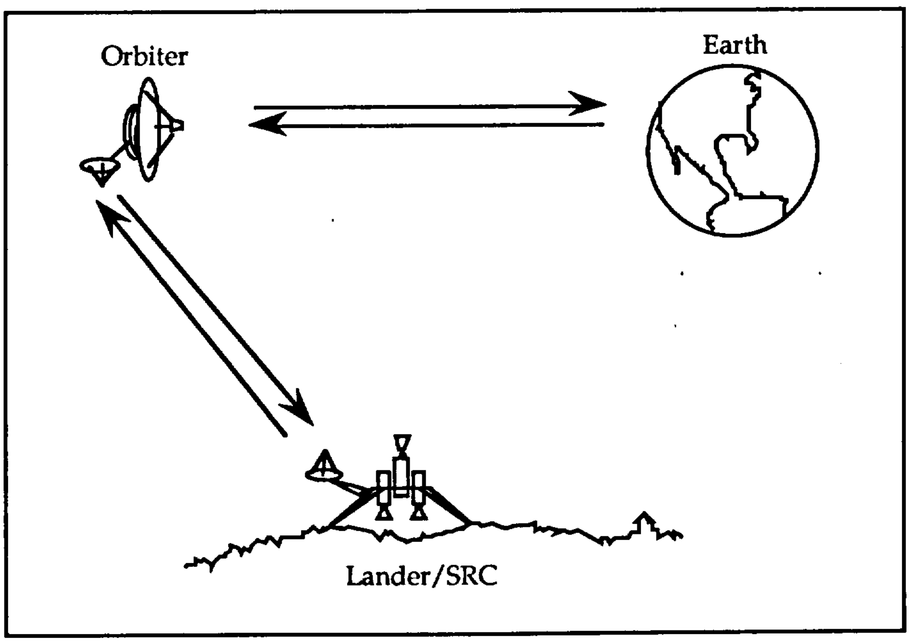


Figure 16: Orbiter Relay Communications Architecture.

5.4.2 High Gain Antenna

The parameters for the orbiter HGA are outlined in Table 12. Operating at a frequency of 8420.43 MHz with a radio frequency (RF) power of 20 W, the 4 m-diameter high gain antenna provides a transmission gain of 49.08 dB. This is sufficient to provide a minimum transmission data rate of 12,680 bps at a maximum range of 3.0083 AU. This data rate increases as the orbiter nears the Earth, as shown graphically in Figure 17.

Also displayed is the change in space loss for the HGA, shown in Figure 18. The space loss is compensated for by using a large gain or a high radio frequency transmission power. The space loss decreases as the transmission distance decreases, thereby allowing an increase in transmission data rate for a fixed gain and RF power.

Table 12: Parameters for the Orbiter High Gain Antenna

Parameter	Value
Diameter	4.0 m
Frequency	8420.43 MHz
Gain	49.08 dB
Maximum Transmission Distance	3.0083 AU
Minimum Data Rate	12.7 kbps
RF Power	20 W

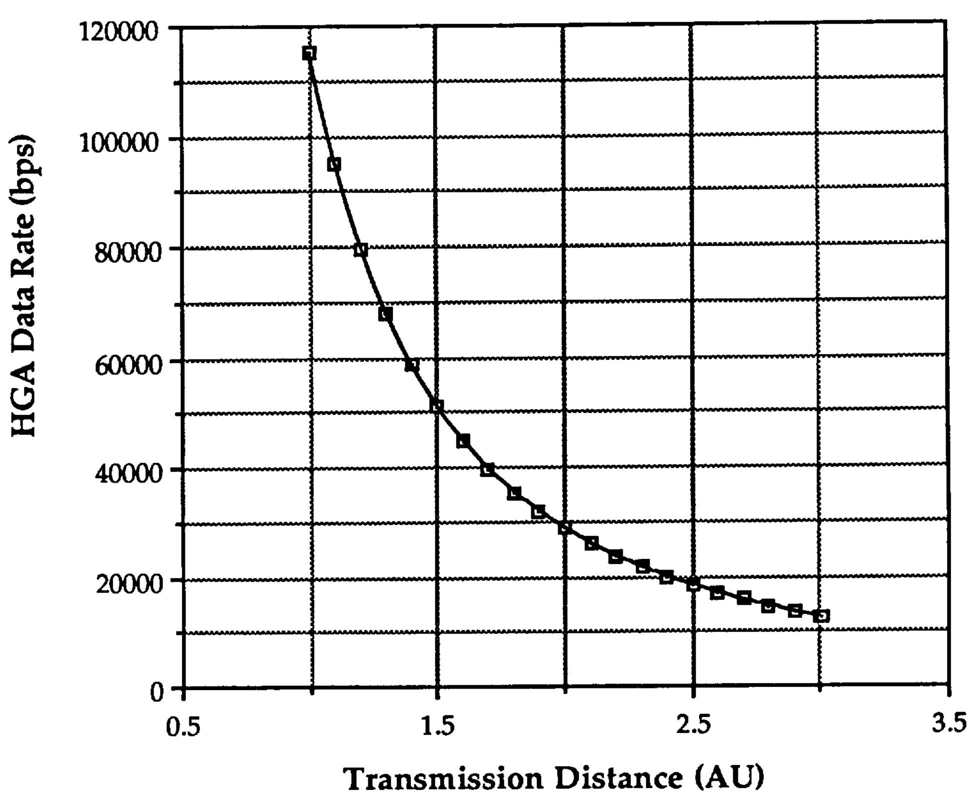


Figure 17: Change in Orbiter HGA Data Rate

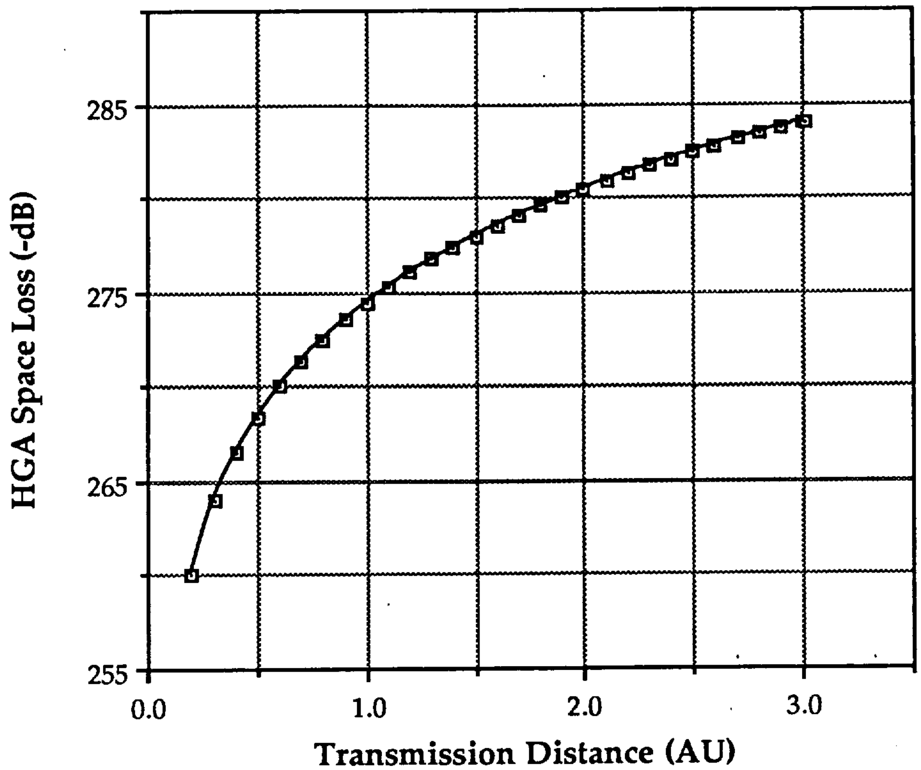


Figure 18: Variation in Orbiter HGA Transmission Space Loss

5.4.3 Low Gain Antenna

Design parameters for the low gain relay antenna are outlined in Table 13. The relay antenna operates at a frequency of 2114.68 MHz with an RF power of 5 W. Since the average transmission distance is 50 km, the relay antenna is 0.365 m in diameter and provides an average data rate of 136,440 bps. This data rate does not vary much because the transmission distance will be maintained at 50 km.

Communication design parameters not displayed here can be found in the orbiter link design table in Appendix D.

Table 13: Parameters for the Orbiter Low Gain Relay Antenna

Parameter	Value
Diameter	0.365 m
Frequency	2114.68 MHz
Gain	16.27 dB
Average Transmission Distance	50.0 km
Average Data Rate	136,440.0 bps
Radio Frequency Power	5.0 W

5.5 Guidance, Navigation and Control

The guidance, navigation and control (GNC) subsystem is an integral part of the entire mission scenario (see Section 4.1). Since the orbiter uses Mariner Mark II components, a similar GNC system is incorporated into Hawking. Furthermore, large communication lag times require the spacecraft's navigation system to be semi-autonomous.

5.5.1 Orbiter GNC Requirements for Mission Phases

After departing LEO, the Hawking spacecraft performs inertial and velocity measurements to provide attitude references for both large trajectory transfer burns and TCM's. An inertial reference, required by the on-board navigation system, is provided by a sun sensor, a star tracker, a horizon sensor, and an Earth sensor. The Earth sensor is a star tracker used to track the Earth. Deviations from the desired attitude are corrected with helium cold-gas jets and momentum wheels.

During phase 4, the spacecraft inserts into the asteroid's heliocentric orbit and begins to map the surface. These maneuvers require a high degree of pointing accuracy, as well as slewing (rotation about the center of mass) of the spacecraft. During the mapping maneuvers of phase 4, nominal rates of slewing (between

0.05 to 0.5 deg/s) are controlled with the lander's OF_2/B_2H_6 thrusters and helium cold-gas jets.

After the lander/SRC docks with the asteroid, the orbiter remains at a specified distance from the asteroid to act as a communication relay station. GNC components for orbit maintenance include the inertial measurement unit, reaction and momentum wheels, control moment gyros, hydrazine thrusters, and helium cold-gas jets.

5.5.2 Spacecraft Stabilization

The zero-momentum method of three-axis stabilization uses three reaction wheels in unison with helium cold-gas jets to maintain spacecraft stability about all three axes. The reaction wheels and jets provide accurate pointing capability, and three axis stabilization allows any attitude to be achieved without constraint. High pointing accuracy is necessary for communication relay and precision mapping; accuracies as high as 10^{-4} degrees can be achieved.

5.5.3 Sensors and GNC Actuators

Establishment of the GNC requirements necessitates a study of the various mechanisms used to satisfy them. Available control actuators which are presented in Table 14 with their performance range, weight, and power characteristics.

Four sets of thruster packages (15-18 N per thruster) are placed at 90 degree increments around the circumference of the spacecraft's hub. Each package includes a set of two thrusters pointing opposite one another to control spin and a similar set to control pitch and yaw. This provides full three-dimensional translational and rotational control. The orbiter uses hydrazine for the large thrusters. For highly accurate pointing maneuvers, helium cold-gas jets producing <0.2 N of thrust are placed in the same configuration as the large thrusters.

Table 14: Control Actuators.(3)

Actuator	Typical Performance Range	Mass (kg)	Power (W)
Hydrazine Thrusters	15-18 N	0.25	N/A
Helium cold-gas jets	0.1 - 0.2 N	0.15	N/A
Reaction Wheels	0.4 to 400 N-m-s at 1200 to 5000 rpm	6.0	11

In addition to attitude control and stabilization, the spacecraft must determine its initial attitude and obtain an inertial fix before performing maneuvers. The sensors required to perform these tasks are presented in Table 15.

Table 15: GNC Sensors (3)

Sensor	Typical Performance Range	Mass (kg)	Power (W)
Inertial Measurement Unit: Gyros & Accelerometers	Gyro drift rate=0.003 to 1 deg/hr Accel. linearity=1 to 5×10^{-6} g/g^2	20	25
Sun Sensor	Accuracy=0.005 to 3 deg	4 - 5	6 - 8
Star (Canopus and Earth) Trackers	Attitude accuracy=1 to 60 arc sec	5 - 8	8 - 10
Scanning Horizon Sensor	Attitude accuracy=0.1 to 1 deg	4 - 5	6 - 10

5.6 Structures and Thermal Control

The main structural component of the orbiter vehicle is the 12 sided annular ring. This ring houses delicate computer and scientific equipment and serves as the mounting platform for the high gain antenna and all booms (high precision scan platform, low precision scan platform, RTG platform, and magnetometer boom). In addition, an aluminum alloy truss structure is mounted to the base of the ring to hold the orbiter's attitude correction thrusters away from the sensitive instruments on the upper parts of the spacecraft.

Thermal control for the orbiter protects sensitive instruments by maintaining them within their respective operating temperature ranges (see Table 16). Since the orbiter contains many of the instruments and sensors, its thermal control system is the most extensive of the three spacecraft component vehicles. All exposed instruments (those not contained inside the annular ring) are covered with protective material coatings such as white paint on the antennas and reflective gold foil and insulation wrapped around instrument packages. In addition, sun shades cover instrument packages located on the various booms, while radiator fins and louvers are used to eject internally generated heat from various instruments. Finally, thermal shields are placed between the annular ring and sources of heat and radiation such as RTG's or thrusters. Passive thermal control devices are used extensively on the orbiter because of their high reliability and low mass.

Table 16: Operating Temperature Range.(4)

Subsystem	Operating Temperature Range (°C)
Structures	-115 / +65
Electronic	0 / +40
Batteries	+5 / +20
Solar Panels (SRC)	-100 / +100
Fuel	-350 / -190
Oxidizer	-240 / -75
Helium Tank	-18 / +43
Navigation Sensors	-30 / +50
Antenna	-170 / +90

5.7 Orbiter Mass Allocation

The masses of the orbiter's subsystems were allocated based on the CRAF/CASSINI Mass and Power Report by JPL. The mass allocations were estimated by averaging the weight of the CRAF and CASSINI missions and then weighting each value based on the design of the orbiter section of the Hawking (see Table 17).[5]

Table 17: Mass Allocation for the Orbiter.[5]

Subsystem	Mass (kg)
Structure	200.0
Radio Frequency	35.0
Power and Pyro	120.0
Command and Data	30.0
Attitude and Articulation Control	50.0
Cabling	50.0
Propulsion Module Hardware	50.0
Thermal Control	60.0
Mechanical Devices	40.0
Digital Tape Recorder	34.0
Antenna	21.0
Science Instrument	11.0
Spacecraft Purge Equipment	1.4
Total Mass	902.4

6.0 The Lander

The lander is a new design but will use advanced light weight composites as part of its structure and tanks. One idea is to adapt the common lunar lander (CLL) structure as the base for the Hawking lander. The development of the CLL is scheduled for completion in 1995/96 and using their research could save development time and costs of the Hawking lander

6.1 Lander Mission Objectives

The mission objectives of the lander are to:

- Provide a mounting structure for the attitude thrusters that control the spacecraft during the initial cruise and asteroid landing.
- Carry the tanks for the fuel used during the trajectory correction maneuver, insertion into the asteroid's orbit and asteroid docking.
- Provide a safe landing and launch pad for the SRC.
- Send data to the orbiter to update the status of the SRC or science experiments.
- Gather samples and place them inside the SRC.
- Provide power and a platform for extra science experiments.

6.2 Scientific Instruments

The main mechanical systems present on the lander are the micro rovers, the robotic arm, and coring mechanism. These devices will be used to retrieve samples of material for transport back to the earth. Because of the electrical power demands of these devices an RTG will be placed on the lander. To utilize this long-lived power source after the SRC's departure, 20 kg has been reserved on the lander for scientific instruments. Some of the instruments that could be carried on the lander are listed in Table 18, along with their function.

Table 18. Lander's Scientific Instruments and their Function.

Function	Instrument
Surface examination	Seismic Detector
	Mass Spectrometer
	Surface Scraper / Compacter
	Extra 20 kg of scientific instruments
Surface observation	Imaging telescope
Sample acquisition	Core Borer
	Robotic Arm
	Micro-Rover

Of particular interest is the capability to perform analysis of samples *in situ*. This would ensure the return of valuable data in the event a sample was lost on route to LEO.(1)

6.3 Power

The lander power system has multi-role requirements:

- Supply the SRC with power during the cruise to the asteroid and its stay on the asteroid
- Supply power to short duration, high loads (such as the coring mechanism)
- Supply power to scientific experiments for long durations.

To meet the above requirements, a single RTG which supplies 284 W of power and weighs about 56 kg.(2,3) was chosen for the power system. Solar panels were not used because of concern over lack of light or dust clouds. Fuel cells were also eliminated because their limited lifetime and weight.(3)

6.4 Communications

The lander/SRC combination will dock on the asteroid and utilize the low gain antenna (LGA) on the lander to communicate with the orbiter. The orbiter will relay this information received from the lander to Earth and will also relay command information to the lander/SRC (see Figure 16). The lander will continue to utilize this data relay configuration to communicate with the Earth even after the SRC has departed the asteroid.

The lander's low gain relay antenna is essentially the same as the relay antenna of the orbiter (Table 13). Design parameters for the lander communications systems not displayed here may be found in the complete link design table in Appendix D.

6.5 Structures and Thermal Control

The main structure of the lander is the truss-like frame which supports the SRC during the docking phase of the mission and holds the SRC to the asteroid surface. The frame also serves as a mounting structure for the thrusters, propellant and pressurant tanks, and science instruments which remain on the asteroid surface.

Thermal control for the lander is all passive. The structure is covered with protective material coatings (white or gray paint) designed to reject incoming radiation. As with the orbiter, foil and insulation are used to protect instruments (such as the mechanical arm package) and maintain them within their operating temperature ranges. Thermal shields are placed around the thrusters to protect the lander/SRC from heat.

6.6 Lander Mass Allocation

The masses of the lander's subsystems were allocated based on the CRAF/CASSINI Mass and Power Report by JPL and the masses of other lander vehicles (Viking and Luna). The mass allocations were estimated by averaging the subsystem masses of the CRAF and CASSINI missions and then weighting each value based on the design of the Hawking lander and other lander vehicles (see Table 19).[4]

Table 19. Mass Allocation for Lander Subsystems.[4]

Subsystem	Mass (kg)
Structure	58.2
Radio Frequency	11.0
Power and Pyro	66.7
Command and Data	8.0
Attitude and Articulation Control	0.0
Cabling	10.0
Propulsion Module Hardware	186.7
Thermal Control	0.0
Mechanical Devices	60.0
Digital Tape Recorder	17.0
Antenna	5.0
Science Instrument	20.0
Spacecraft Purge Equipment	0.0
Total Mass	442.6

7.0 The Sample Return Craft

The SRC is essentially an enlarged "brilliant pebble" that will carry the sample back to Earth orbit. Development time and cost of the SRC could be reduced by using the miniaturization, materials and propulsion technology created for the "brilliant pebbles" program.

7.1 SRC Mission Objectives

The mission objectives of the SRC are to:

- Dock the lander to the asteroid.
- Return the sample safely to an Earth orbit.
- Protect the sample on the return flight to the Earth.

7.2 Power

During the cruise to the asteroid, insertion and asteroid docking, the power of the SRC is supplied by the lander's RTG. However, once the SRC is ejected from the lander, it has to supply its own power.

The primary requirement on the SRC power system was to supply 50 W of power from a mass of only 10 kg. The length of the mission eliminated fuel cells while standard RTG's were left out because of their weight and their potential to harm the environment upon arrival at Earth (if their was a mishap).

Solar panels, working in conjunction with batteries, are able to supply the necessary power and meet the mass constraint. Assuming that at least 45% of the cells will be in the sunlight at any given time, the solar panels need 1.28 m^2 to supply the required power. The solar cells are the k6(3/4), 10 ohm-Cm type, 8 mil thick and have a 6 mil coverglass. If the power from the solar panels dips below 50 W, the batteries can cover the difference.(1)

Nickel-cadmium batteries were chosen because their energy density (25-30 W-hr/kg) was enough for the mission requirements and they have been space tested. Four 4 cells at 20 Ah each are required.

The batteries are charged by the lander's RTG at all times during the mission. The batteries supplies the SRC with power during the initial launch from the lander and during the transfer trajectory burn back to the Earth. Once the boosters of the SRC are jettisoned, the solar panels are deployed and augment the batteries. The total weight of the solar panels is 5.8 kg and the total weight of the batteries is 2.2 kg.(1)

7.3 Communications

While the SRC is still attached to the lander on the asteroid's surface, it uses the lander's LGA to communicate to Earth via the orbiter. After separating from the lander, the SRC communicates directly with the Earth over a LGA since only low volume data (command and status) is transmitted to and from the SRC. Parametric values for the SRC low gain antenna are in Table 20.

Table 20: Parameters for the SRC Low Gain Antenna

Parameter	Value
Diameter	0.355 m
Frequency	8435.14 MHz
Gain	28.06 dB
Maximum Transmission Distance	3.0083 AU
Minimum Data Rate	50 bps
Radio Frequency Power	5 W

As with the orbiter high gain antenna, the SRC's LGA data rate will increase as the SRC nears the Earth, attaining 11,312 bps at a range of 0.2 AU, while the space loss decreases to approximately -260.5 dB (see Figures 20 and 21, respectively).

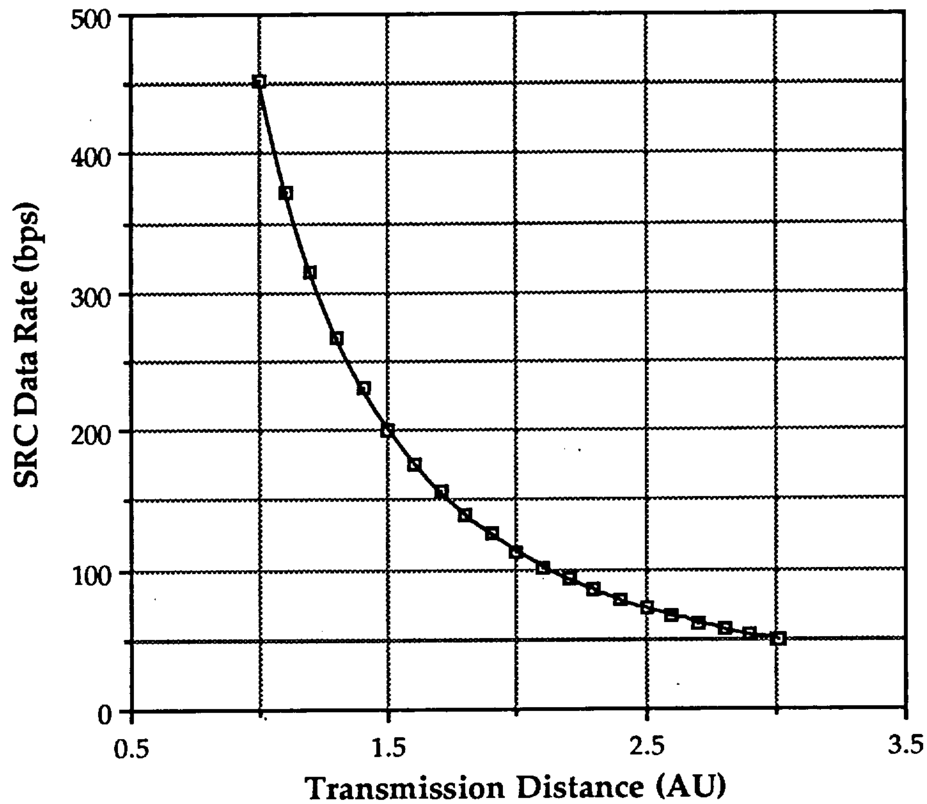


Figure 19: Variation in SRC Low Gain Antenna Data Rate

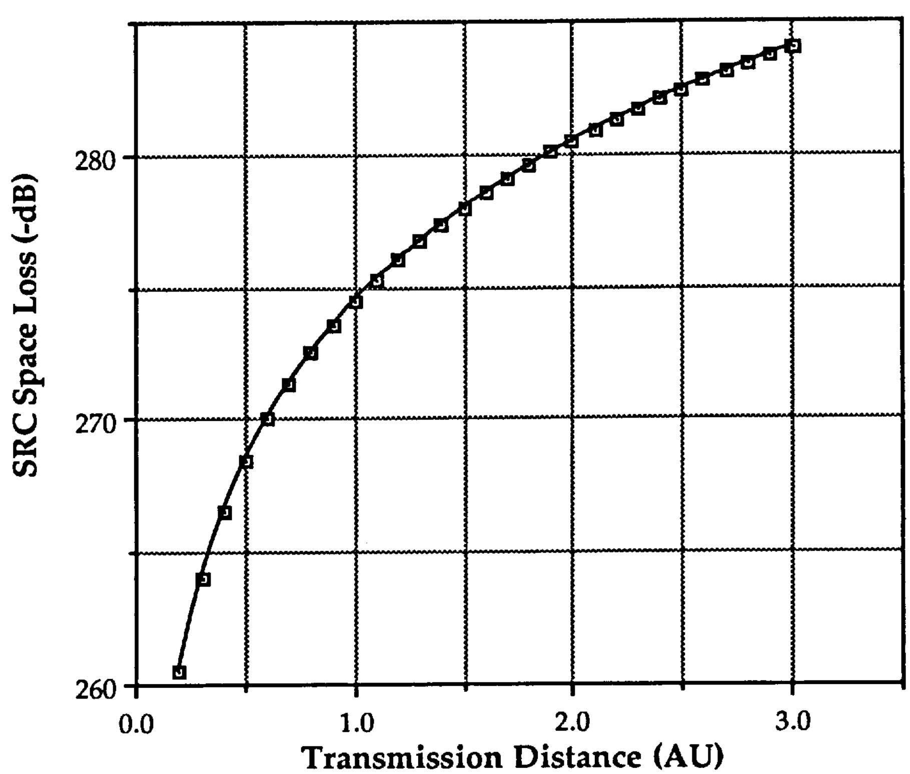


Figure 20: Change in SRC Transmission Space Loss

Design parameters for the SRC communications systems not displayed here may be found in the complete link design table in Appendix D.

7.4 Guidance, Navigation and Control

Upon separation from the orbiter, the SRC uses its own GNC system during asteroid approach. As with the orbiter, the communication time lag with Earth makes it necessary for the SRC's navigation subsystem to be semi-autonomous. The functions which the GNC subsystem performs throughout the mission are outlined in this section.

7.4.1 Spacecraft Stabilization

The total mass of the GNC system was limited to 10.7 kg. For this reason it is desirable to use a 3-axis stabilization method employing thrusters and one double-gimbaled momentum wheel. This system of stabilization is referred to as a bias-momentum 3-axis stabilization method. The momentum system using a single momentum wheel is much lighter than the three wheel zero-momentum configuration. The characteristics of this method, including pointing options, maneuverability, accuracy, and lifetime, are listed in Table 21.

Table 21. Bias Momentum Stabilization Method (2)

Type	Bias Momentum (1 double-gimbaled wheel & roll thrusters)
Pointing Options	Best suited for local vertical pointing
Translation Maneuver	Same as zero momentum with full set of thrusters; otherwise, not suited to translation
Rotation Maneuver	Momentum vector of the bias wheel prefers to stay normal to orbit plane, constraining yaw maneuver
Accuracy	Depends on sensors but generally less accurate than zero momentum
Lifetime	Propellent; Life of sensor bearings

7.4.2 Actuators and Sensors

Table 22 summarizes the performance range, weight, and power requirements of the control actuators. In particular, it should be noted that oxygen difluoride is used as the propellant for the 15 N thrusters. This allows the use of one set of SRC propellant tanks, providing additional savings in mass.

Table 22: Control Actuators (2)

Actuator	Typical Performance Range	Mass (kg)	Power (W)
Oxygen Fluoride Thrusters	15 N	0.25	N/A
Helium cold-gas jets	0.1 - 0.2 N	0.15	N/A
Reaction Wheel	0.4 to 400 N-m-s at 1200-5000 rpm	2.00	11

Before performing any maneuvers, it is necessary to determine the craft's initial attitude and inertial fix. This is accomplished through the use of sun sensors, star trackers, horizon sensors, and an inertial measurement unit. These GNC sensors and their performance ranges, weights, and power requirements are detailed in Table 23. The masses shown in Table 23 represent values that satisfy the SRC mass limit on the GNC subsystem.

Table 23. GNC Sensors for the SRC.(2)

Sensor	Typical Performance Range	Mass (kg)	Power (W)
Inertial Measurement Unit: Gyros & Accelerometers	Gyro drift rate=0.003 to 1 deg/hr Accel. linearity=1 to 5×10^{-6} g/g^2	<3	25
Sun Sensor	Accuracy=0.005 to 3 deg	<1	6 - 8
Star (Canopus and Earth) Trackers	Attitude accuracy=1 to 60 arc sec	<2	8 - 10
Scanning Horizon Sensor	Attitude accuracy=0.1 to 1 deg	<1	6 - 10

7.4.3 GNC Thrusters

For maneuvering, sets of thruster packages will be placed at 90 degree increments around the circumference of the SRC. Each package includes two thrusters pointing in opposite directions to control spin, and two more to control pitch and yaw. This provides full three-dimensional translational and rotational control. The SRC must use oxygen-difluoride as the propellant for the large thrusters to avoid adding additional tank mass. The mass of each of these thrusters is approximately 0.2 kg.

For highly accurate position corrections, as required for pointing maneuvers, helium cold-gas jets producing less than 0.3 N of thrust will be placed in the same configuration. Lightweight (0.15 kg) models of these jets are commercially available.

7.5 Structures and Thermal Control

Due to mass constraints, the structure of the SRC has been minimized. Basically, the SRC consists of the sample container mounted on a configuration of propellant and pressurant tanks and thrusters. The only significant structure is in the connection of the main boosters to the sides of the SRC and the mounting structure for the solar arrays.

Thermal controls for the SRC component vehicle are passive and minimal due to mass constraints. Without extensive instrumentation, the only thermal concern is maintaining the sample temperature close to the temperature present on the asteroid to retain the sample's integrity. To accomplish this task, thermal shields are placed between the sample container and the boosters. In addition, the SRC utilizes a "barbeque-roll" technique during the return to LEO cruise to insure an even temperature distribution through out the sample.

7.6 SRC Mass Allocation

The masses of the SRC's subsystem were allocated based on the CRAF/CASSINI Mass and Power Report by JPL and the known masses of a "brilliant pebble". The mass allocations were estimated by averaging the subsystem masses of the CRAF and CASSINI missions and then weighting each value based on the design of the SRC and "brilliant pebbles" (see Table 24).

Table 24: Mass Allocation for the SRC Subsystems (3).

Subsystem	Mass (kg)
Structure	12.0
Radio Frequency	1.8
Power and Pyro	12.0
Command and Data	2.0
Attitude and Articulation Control	10.7
Cabling	3.0
Propulsion Module Hardware	13.0
Thermal Control	5.0
Mechanical Devices	0.0
Digital Tape Recorder	0.0
Antenna	3.0
Science Instrument	0.0
Spacecraft Purge Equipment	0.5
Sample of Asteroidal Material	3.0
Total Mass	66.0

8.0 Contingency Plans

Although missions are suppose to go off without problems, contingencies are always necessary. Table 25 is a list of possible failures and corresponding countermeasures.

Table 25. Contingency Plans During Various Phases of the Mission.

Phase	Event	Consequences	Countermeasures
1	Propellant spill	Kill/maim launch technicians	Strict safety precautions.
	Explosion on pad	RTG's rupture and corrosive propellants atomize	Launch with seaward wind to minimize causalities
	Explosion during ascent	RTG's rupture and corrosive propellants atomize	Launch with seaward wind, and groundtrack over the ocean
	Spacecraft & upper stage fail to achieve LEO	RTG's drop to Earth and propellant tanks rupture	Explode propellant tanks at high altitude, burning most of the fuel. Jettison RTG's to freefall into Atlantic safety zone
2	Upper stage fails to fire or fires incompletely.	Spacecraft fails to achieve proper injection	Use propellant on Hawking for injection into a nuclear-safe orbit
3	GNC of the orbiter fails to control the spacecraft	Unable to point Hawking to communicate or for TCM's	SRC GNC able to take over for orbiter if necessary
	TCM maneuver misfire	Asteroid aimpoint missed	Perform several TCM's for redundancy
4	Misfire during insertion at the asteroid	Spacecraft executes a flyby of the asteroid	Redundancy in the fuel system and in the insertion engines controls
	Imaging system failure	Spacecraft unable to map the asteroid	Redundant imaging systems
5	Failure during SRC/lander separation from the orbiter	SRC/lander do not separate from the orbiter	Redundant separation systems
	Catastrophic failure during landing	SRC/lander crash or tumble out of control	Provide a safe-mode for the spacecraft during landing

6	Failure in sample collection mechanisms	Cannot retrieve samples	Alternate routes of operation for the mechanism, multiple modes of sample collection
7	Failure of non-destructive launch mechanism for the SRC	SRC can not launch from the lander	Redundant separation mechanisms, fire SRC boosters as a last resort
8	Boosters on the SRC misfire	SRC tumbles or misses injection	Safe-mode and A.I. in GNC system Redundancy in propulsion system
	Boosters fail to separate from the SRC	Solar panels can not deploy and sample lost due to power failure	Redundancy in separation system. Design of SRC that enables deployment of solar panels with boosters attached
9	Misfire during Earth insertion	Loss of sample in an Earth flyby	Redundancy in propulsion systems
	Misfire during circularization burn (LEO)	Sample stuck in highly eccentric orbit	Redundancy in propulsion system Possible mission to send an interceptor craft to bring SRC to LEO

9.0 Recommendations for Further Mission Development

NEW WORLDS INC. considers the following recommendations for NASA to be of primary importance; the list is by no means complete. Suggestions that concern specific aspects of Hawking and its mission come first, because they require immediate attention. Next are recommendations which address more supplemental aspects of the mission.

Study potential opportunities to perform a planetary flyby, thereby reducing the propellant required for the mission. Develop a more accurate Lambert Targeting optimization code to refine the ΔV's.

Analyze the power required by Hawking during all mission phases.

Model Hawking's booms and truss structure with MSC/NASTRAN. Transient and steady-state responses produced by pulsed and steady-state thrust, as well as the spacecraft's natural modes, are of primary concern.

Research alternate high performance propellants that are less toxic and corrosive. Determine effectiveness of teflon and inconel to resist corrosion. Research alternate lightweight materials with similar or better corrosion-resistant properties than teflon and inconel.

Evaluate the potential to adapt Brilliant Pebbles and ALAS technology to the Hawking spacecraft--especially the SRC.

Study the feasibility of adapting the Common Lunar Lander (CLL) concept to the Hawking lander in order to reduce cost and development time of the spacecraft.

Perform detailed investigations of the candidate asteroids with the Hubble Space Telescope.

Pursue extra asteroid encounters during outbound cruise to the asteroid in order to increase the scientific return of the mission.

Assess the potential capability of Space Station Freedom to analyze the returned sample as opposed to performing sample analysis on Earth.

10.0 Organization

This section describes the company structure of NEW WORLDS, Inc., the responsibilities of each team, and a cost breakdown of the project.

10.1 Organization Structure

The organization of NEW WORLDS, Inc. is shown in Figure 21. The specific responsibilities, members and leaders of each team are explained in Section 9.3. However, the many teams are part of two basic groups.

The first group consists of the Asteroid Utilization/Mission Design who are responsible for analyzing the costs and benefits of each scenario, justifying the mission and integrating of the mission into future space exploration plans.

The second group consists of the Spacecraft and Subsystem Group who are responsible for determining how to successfully execute each scenario. They are involved with analyzing spacecraft systems of now and the future.

The Mission and Spacecraft Review Group consists of all members and provides a forum for review and integration.

10.2 Integration

The key to reducing the number of problems of this project is the Mission and Spacecraft Review Group which contains all the members of NEW WORLDS, Inc. Here ideas are exchanged in an open forum, problems are found and resolved, and suggestions and improvements are incorporated into designs. This arrangement also ensures that integration between the subgroup will be a continuous process instead of a sudden shock at some later date.

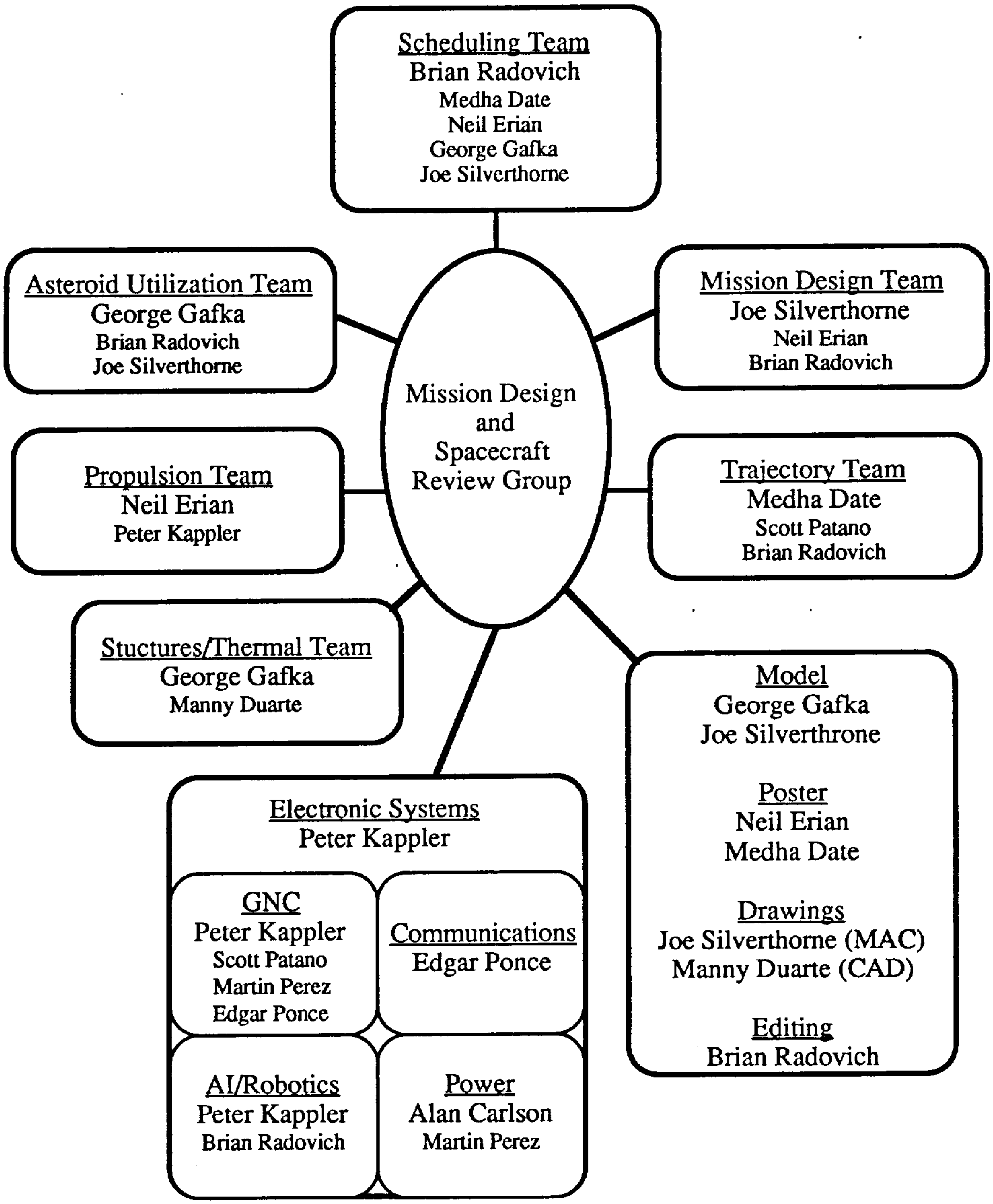


Figure 21: Organizational chart of NEW WORLDS, Inc.

10.3 Responsibilities of Each Team

Each of the teams is responsible for certain areas of the mission or spacecraft. These responsibilities are illustrated in Table 26:

Table 26. Team Responsibilities.

Team	Responsibilities
Scheduling	Creates schedules, time lines and task charts
Asteroid Utilization	Investigates utilization of asteroids (mining, etc ...)
Mission Design	Creates exploration and utilization scenarios, selects scientific instruments, creates contingencies
Propulsion	Evaluates current/near future propulsion capabilities for exploration mission, determines mass of the spacecraft during the mission
Trajectory	Finds asteroids with low ΔV's, determines launch windows and trajectory paths
Structures/Thermal	construction and integration of spacecraft components, modularity
Electronic Systems	Integrates spacecraft components during the various mission phases
GNC	Determines components necessary to control the attitude of the spacecraft
Communications	Determines necessary communication equipment to perform the mission
AI/Robotics	Researches pace of AI/Robotic development, determines requirements for this mission
Power	Determines the power requirements, finds the necessarily equipment
Special Tasks	
Model	Build a model of the Hawking spacecraft
Poster	Construct a poster to explain the Hawking mission
Drawings	Draw the Hawking spacecraft
Editing	Integrate the document from various authors

10.4 Cost of this Project

This section contains estimated costs for personnel, material and hardware necessary to complete this project.

Personnel Cost Estimates

Personnel	Hourly Wages	Weekly Hours	Total Salary
1 Project Manager	$ 30.00	10 hrs/wk	$ 300.00
7 Team Managers	22.00	3 hrs/wk	462.00
11 Engineers	17.00	15 hrs/wk	2,805.00
Weekly Subtotal			$ 3,567.00
Total Personnel Cost for 14 Weeks			$49,938.00

Material and Hardware Cost Estimates

Material/Hardware	Fourteen Week Total
3 Macintosh IIsi Computers/software/etc.	$ 1,400.00
1 IBM PC/software/peripherals	600.00
Photocopies ($0.05 each)	15.00
Transparencies ($.25 each)	40.00
MODEL AND POSTER (real cost, no estimate)	200.00
Miscellaneous (Gas, phone, etc...)	20.00
Total Estimate	$ 2,375.00

Total Estimated Cost

Personnel Cost Estimate	$ 49,938.00
Total Estimate	$ 2,375.00
Total 14 Week Cost Estimate	$ 52,313.00

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Appendix A: Asteroid Utilization

A.1 Background and Utilization 71
A.1.1 Mining 72
A.1.2 Scientific Research 72
A.1.3 Spaceport 72

A.2 Scenario I: Asteroid Retrieval 73
A.2.1 A Word on Moving an Asteroid 73
A.2.2 Advantages and Disadvantages of Scenario I 74
A.2.3 Asteroid Selection Criteria for Scenario I 74

A.3 Scenario II: Asteroid Mining/Refuelling Waystation 75
A.3.1 Advantages and Disadvantages of Scenario II 75
A.3.2 Asteroid Selection Criteria for Scenario II 76

A.4 Introduction to Advanced Propulsion Concepts 76
A.4.1 Chemical Rockets 77
A.4.2 Electric Propulsion 77
A.4.3 Nuclear Pulsed Propulsion 79
A.4.4 Nuclear Thermal Rockets 79
A.4.5 Antimatter Drives 80
A.4.6 Propulsion System Evaluation 80

A.5 References 81

A.6 FORTRAN program SPIRAL.FOR 82

A.1 Background and Utilization

Utilization of space-based resources is essential to future space exploration and colonization efforts. Manufacturing components on the Earth (using terrestrial materials) and assembling them in orbit has limited application when massive or bulky structures are needed. Therefore, it is not practical to completely support either deep-space or lunar activities with materials supplied from the Earth. For effective exploration and eventual colonization of the solar system, the natural resources present in space must be exploited to the fullest.(1)

Additional impetus for utilization of these resources is brought about by the Earth's finite amount of exploitable natural resources which are being depleted at increasing rates. Much time and effort is spent in the quest for new deposits of natural resources (oil, natural gas, metals, etc.), however demand outweighs supply and many sources of high grade ore are near exhaustion. As the high grade ores are depleted, mankind is forced to use lower grades. Lower grade ores require more energy to process and produce greater amount of waste products than the high grade ores. In addition, the waste produced is often more environmentally damaging than the waste from a higher grade ore source. The effects of generations of environmental neglect and abuse are now becoming evident.(2)

Utilization of space-based resources could help alleviate many of the environmental problems created by mineral extraction techniques here on Earth and provide the resource base needed for exploration and colonization of the solar system. One of the many deposits of natural resources in space are asteroids, which may contain billions of tons of valuable materials (iron, nickel, water, etc.).(3)

In short, the utilization of space-based resources such as asteroid could reduce the amount of material launched off the Earth and provide an ample supply of raw materials for a mineral depleted planet. NEW WORLDS, Inc. has undertaken the task of investigating ways in which the natural resources present in asteroids can be utilized to benefit both the Earth and activities of man in space.

A.1.1 Mining

Asteroids are classified depending upon composition. Each category is rich with various raw materials and useful gases. One particularly promising category is chondritic asteroids. It is believed that these have not undergone differentiation and therefore contain vast deposits of nickel, gold, silver and other useful metals near the surface. Along with metals, some asteroids are believed to have ice deposits which could be molecularly separated into oxygen and hydrogen using solar electrolysis.

Development of asteroid resources could available make ample amounts of materials for space construction, life support systems, and rocket propellant. Also, metals might have a crystal structure more pure than those found on Earth due to the micro-gravitational field of asteroids. The tremendous benefit of mining in space is that it eliminates the need to lift materials off the Earth or Moon.

Extracting the metal ore could be done using the standard carbonyl process, a current method of metal purification. The entire refining process requires little power, passive thermal control, and few moving parts. The necessary carbon monoxide could be extracted by heating the asteroidal soil. Sulfur, a catalyst known to accelerate this process, is believed to be abundant on asteroids.(4)

A.1.2 Scientific Research

It is believed that asteroids contain clues about the formation of the solar system that were erased on the larger planetary bodies. The relatively small size of asteroids excluded them from experiencing the evolutionary processes that altered the original structure of the planets and large moons. Therefore, it is suspected that asteroids are composed of primitive matter typical of the region in which they formed. Also, asteroids may contain evidence of long-term fluctuations in the solar wind, micrometeors, solar-flare particles, and galactic cosmic rays. A thorough study of asteroids could vastly increase the knowledge of the birth and evolution of the solar system.(5)

A.1.3 Spaceport

If an asteroid of reasonable size (approximately 1 kilometer) were brought to an Earth orbit, it could be developed into a "spaceport", similar to present day seaports. The spaceport would consist of a permanently manned "space station" mounted on or inside the asteroid and coexist with the established mining operations. There would be a myriad of uses for such a large space station:

- Refueling waystation for Moon, Mars and interplanetary missions

- Mining to support the expansion of the spaceport, to replace the dwindling resources of Earth and to supply raw materials for manufacturing new spacecraft

- Point for assembly, maintenance and refurbishment of large spacecraft

- Advanced training facility for EVA (Extra Vehicle Activity) missions

The diverse nature of the spaceport along with the increasing necessity for its uses makes it a promising prospect.

A.2 Scenario I: Asteroid Retrieval

In this scenario, a near Earth asteroid is returned to an Earth orbit. The final location is dependent upon whether the mined materials are used to support terrestrial activities or for construction of space structures.

Phase 1: Explorer missions are launched to candidate asteroids. Selection of the asteroid to be returned is based upon the data of the explorer missions. Also at this time, new technologies are developed and verified so that construction on the asteroid propulsion unit (APU), the inter-planetary transfer vehicle (ITV), and the automated mining equipment (AME) can begin.

Phase 2: Construction multiple APU's, AME and ITV components and launch into an Earth orbit. Assembly takes place in Earth orbit.

Phase 3: The ITV (with APU's and AME cargo) is launched into a rendezvous trajectory with the asteroid. Upon arrival, multiple APUs a are docked with the asteroid and the asteroid is despun. The AME units are also unloaded and mining begins. The ITV is attached to the asteroid. The APU's are realigned and begin nudging the asteroid towards Earth. Below in Section A.2.1 is a more detailed discussion about moving asteroids.

Phase 4: The asteroid is mined during its entire journey to Earth (approximately 10yrs). As the asteroid nears the Earth (about 1 year prior to arrival), additional APU's are sent to aid in the final orbit insertion. The approach of the asteroid is targeted to aid its placement into the desired Earth orbit.

Phase 5: After the asteroid is inserted into the desired Earth orbit, the APUs are removed and development of the asteroid continues. Additional AME units are placed on the asteroid and final refining of materials is begun. Also, the construction of a spaceport begins.

A.2.1 A Word on Moving an Asteroid

The problem of transferring an asteroid into an Earth orbit is one of mass. A 500m diameter asteroid has a mass of approximately 10^8kg while a 2000m diameter is 10^{13}kg. Because the mass being moved is so high, the APUs change the velocity of the asteroid only in very small increments (acceleration ~ 10^{-5}g). Thus the asteroid would be slowly nudged by the propulsion system along a spiral trajectory over a period of several years.

To assess the magnitude of the ΔV for the return portion of the mission, a FORTRAN program models the spiral trajectory as a series of Hohmann transfers (see Section A.7). This model accounts for the ΔV required for capture at Earth but excludes the plane changes because it uses Lambert targeting for the final approach to Earth. The capture at Earth is modelled using the patched conics method.

A program that can model the application of thrust during the entire flight is needed for more accurate ΔV's. A program that uses an iterative Lambert targeting method might be able to properly model the spiral trajectory.

A.2.2 Advantages and Disadvantages of Scenario I

There are many advantages of this scenario:

- A Spaceport (see above) in Earth orbit.
- Flexibility -- utilization of the asteroid can easily change to reflect Earth's needs or technological advances since the asteroid orbits the it.
- Almost limitless supply of undamaged asteroid material for scientific study.
- Accessibility -- once the asteroid is in Earth orbit, travel time to it is short.

Some disadvantages are:

- Massive amounts of equipment must be lifted into space including the ITV, the APU's and the AME's. This project would be analogous in magnitude to the Normandy landings in France (largest military operation in history).
- Complex mission requiring many years to develop the necessary enabling technologies and many more years to implement.
- If the asteroid hits Earth, considerable destruction could occur. If the asteroid misses insertion and does a hyperbolic flyby, all the equipment and investment is probably lost.
- Propulsion system limits size of asteroid to a maximum of 2km.

A.2.3 Asteroid Selection Criteria for Scenario I

After evaluating the overall mission criteria, limitations and risks for this scenario, the asteroid selection criteria are:

- Near Earth asteroids (ΔV of 7.5 km/s or less).
- Diameters between .5 and 1km (2km max -- propulsion limitation)
- Asteroids with metals and radioactive isotopes .
- Low inclination not required because of Lambert targeting method -- but low eccentricity is preferable.

A.3 Scenario II: Asteroid Mining/Refuelling Waystation

The second scenario is to mine an asteroid in its original orbit for both metals and volatiles. The commodities are taken to the desired location (Earth, Moon or Mars) or used *in situ* by large interplanetary spacecraft (refueling station).

Phase 1: The exploration phase of this mission is the same as the Asteroid Retrieval Scenario. Research and verification testing exclude asteroid propulsion units (APU's).

Phase 2: The ITV (interplanetary transfer vehicle) would be assembled in Earth orbit with support from the Earth, Moon and Space Station Freedom. In the final stages of assembly, the AME (advanced mining equipment) units are launched and attached to the ITV.

Phase 3: The ITV (with AME units) are launched into a rendezvous trajectory with the asteroid. Upon its arrival at the asteroid, the ITV docks with the asteroid and unloads the AME units which begin to process the raw material.

Phase 4: The processed materials (both volatiles and metals) are loaded onto the ITV for transport. The ITV (with the processed materials) are inserted into a trajectory to the desired location (Earth, Moon or Mars depending on the mission). The AME units remain on the asteroid and continue to prepare materials for either the next ITV or interplanetary spacecraft in need of refueling.

A.3.1 Advantages and Disadvantages of Scenario II

Advantages of this scenario are:

- No danger of an asteroid colliding with the Earth.
- Flexibility -- spacecraft can be resupplied on route to further destinations or the materials are returned to the desired location.
- Only the valuable materials mined on the asteroid are moved to the desired location, rather than the entire asteroid.
- Require less initial launch mass than Scenario 1.

Some disadvantages are:

- No near Earth Spaceport.
- Difficulty updating equipment on the asteroid to reflect current needs or technological advances because the asteroid is far away.
- Frequent travel to the asteroid to bring back commodities.
- Complex mission requiring many years to develop the necessary enabling technologies and many more years to implement.

A.3.2 Asteroid Selection Criteria for Scenario II

The asteroid selection criteria for this scenario is:

- Diameter > 1 km, preferably spherical (so orbiting is easier).
- Reasonable rotation rate (for easy docking).
- A ΔV from Earth of 8 km/s or less.
- The asteroid should contain both volatiles and metals.

A.4 Introduction to Advanced Propulsion Concepts

The primary problem of spaceflight is one of propulsion. From the early days of Robert Goddard's liquid rocket experiments and his discovery of electric propulsion to the Saturn V and ion propulsion tests in Earth orbit, scientists and engineers have expanded the field and applications of space propulsion and power.

Two asteroid exploration/utlization concepts that use advanced propulsion systems are:

- An Interplanetary Transfer Vehicle (ITV)
- Asteroid Propulsion Unit (APU)

In the Asteroid Retrieval Scenario, the ITV transfers AME's (automated mining equipment) and APU's to the asteroid. The APU's are deployed on the asteroid and transfer it back to the Earth over a period of several years. For Asteroid Mining/Refuelling Waystation Scenario, the payload consists of AMEs which dock with the asteroid upon arrival and begin mining it.

To exploit the resources of near-Earth asteroids, sufficient propulsion systems are required. Short manned missions to asteroids, as well as long duration unmanned flights including asteroid retrieval missions, necessitate a variety of propulsion options. The purpose of this section is to discuss performance characteristics of five space propulsion systems that are either currently in use by NASA or are in theoretical or advanced development stages. The propulsion systems considered are:

1) Chemical
2) Electric
3) Nuclear Pulsed
4) Nuclear Thermal
5) Antimatter

First, a brief description of the how the propulsion system works is given, followed by specific impulse, thrust, power requirements, and total system masses. Finally, evaluations and comparisons of each system, based upon potential performance in missions to asteroids as well as asteroid retrieval, are made. Also, each propulsion system is categorized as conceptually sound or unsound within the context of asteroid missions.

A.4.1 Chemical Rockets

Liquid (LOX/LH2) engines and solid rocket motors have been the heart of NASA's space exploration effort. They are characterized by high thrust forces and low specific impulses. Chemical rockets are powerful enough to move small payloads short distances. Performance of chemical systems is limited by the internal chemical energy of their propellants. Inefficient combustion of chemical propellants results in low exhaust velocities.(6)

A.4.2 Electric Propulsion

The four types of electric propulsion systems considered are: magnetoplasmadynamic and pulsed inductive thrusters, electrostatic ion thrusters, mass drivers, and rail guns. Contrary to chemical, nuclear, and antimatter systems, all are characterized by high efficiencies and low thrusts. Because they require low propellant masses they can efficiently move large payloads over long distances. Another distinction between electric systems and the other three types is that they do not use the stored energy of the propellant as the primary contribution to specific impulse and thrust. Their I_{sp}'s are only limited, theoretically, by the amount of supplied power.

Two types of electromagnetic thrusters are the magnetoplasmadynamic (MPD) and pulsed inductive thruster (PIT). The MPD ionizes a propellant gas (such as argon) into a plasma as the gas passes through an electric current pulse. This current, passed between a central cathode and annular ring anode, induces a magnetic field. The electric-magnetic field combination creates a Lorentz force that accelerates the plasma through a nozzle.

A PIT provides large amounts of power from massive capacitor storage banks. The power output of these banks is used to accelerate hydrazine or ammonia propellant. The performance and power characteristics of both the PIT and MPD are shown in Table A.1.

Similar to the MPD, the electrostatic ion thruster ionizes the propellant gas. However, the electrons are removed from the ionization region. As the positive ions accelerate from the high voltage ionization region to the low voltage exit plane, the separated electrons are dispersed into the exhaust. This is done so that the exit nozzle does not become positively charged, and produce exhaust deceleration. Performance and power characteristics of ion thrusters are shown in Table A.1. (7)

Table A.1. Performance Characteristics of MPD, PIT, and Ion Engines (7)

Performance Characteristics	Magnetoplasma-dynamic (MPD) Thruster	Pulsed Inductive Thruster (PIT)	Electrostatic Ion Thruster
Input Power (MW)	2.8	.04	.5 - 1.5
Power to Average Thrust Ratio (KW/N)	20 - 60	—	40 - 60
I_{sp} (s)	1000 - 1300	1500 - 2000	6000 - 7000
Thrust (N)	75	1.67	13 - 30
Dry System Mass (kg x 10^2)	—	—	30 - 50
Efficiency (%)	20	35	73
Service Life	Short	10^1+ years	25000 hours

A mass driver is a system that accelerates a given mass to a desired velocity using a travelling magnetic field interacting with a magnetic dipole. A cylindrical bucket with superconducting coils carries a payload, and is guided without physical contact over a system of coaxial drive windings until it reaches its final velocity at the exit plane of the accelerator. Current is provided to the drive windings by large capacitor banks. (8) This type of a system can be used either to launch a usable piece of material such as a tank of oxygen, or it can launch small amounts mass to impart propulsive thrust to the attached body.

Two mass driver launching systems are an electromagnetic launcher (EML) and an electromagnetic launcher using superconductivity technology (quenchgun). These high power accelerators are designed to launch a one metric ton (MT) capsule of lunar produced oxygen into orbit around the moon. Some of the characteristics of each launch system are shown in Table A.4.

Table A.4. Electromagnetic Launcher (EML) Characteristics (9), (10)

System Characteristics	Electromagnetic Launcher (EML)
Length (m)	70 - 150
Mass (metric tons)	300 - 700
Input Power	5 MW, 14.4 Gj per launch
Exit Velocity (m/s)	1700

An important consideration of a high-mass projectile launcher as a means of spacecraft propulsion is its launch rate. The EML has an upper limit of ten launches during an eight hour shift, and the quenchgun can launch five times in five hours. Both systems require a several hour warm down and reset period between work shifts to prepare for the next launch. (9), (10)

Mass drivers that can continuously produce thrust to move a spacecraft are feasible. However their physical and performance characteristics are different from an electromagnetic launcher. Projectile masses on the order of tens of grams are launched at a frequency of 5 Hz from a coaxial system of drive coils that are several kilometers long.(8) The performance and power characteristics of mass drivers for spacecraft propulsion are shown in Table A.3.

Rail guns are similar to mass drivers in that they use a combination of electric and magnetic fields to accelerate projectiles down axial windings. However, the projectile is in physical contact with long rails through which a large current passes. A metallic fuse that interconnects both rails transforms into a plasma as the current passes through the rails. An induced magnetic field perpendicular to the electric field creates a Lorentz force which accelerates the plasma and projectile to the exit plane of the gun. Performance and power characteristics of rail guns are shown in Table A.3. (8)

Table A.3. Physical and Performance Characteristics for Mass Driver and Rail Gun Thruster (8)

Characteristics	Mass Driver	Rail Gun Thruster
Efficiency (%)	>50	25 - 35
Specific Impulse (s)	1000 - 1500	1000 - 2000
Thrust (N)	100 -2200	10 - 100
Input Power (MW)	1 - 25	.1 - 4
Power to Average Thrust Ratio (KW/N)	9 -35	20 -50
Projectile Mass (gm)	1 - 30	.1 - 1
Dry System Mass (kg x 10^5)	1 - 7	.1 - 1
Total Length (km)	12 - 45	20 - 200

A.4.3 Nuclear Pulsed Propulsion

Nuclear pulse propulsion (NPP) is a promising concept which has never been put to practical use. NPP offers a combination of low system mass, high specific impulse, and most importantly, high thrust, which is the dominating criterion in choosing an asteroid propulsion system.

NPP imparts momentum to a vehicle through a series of small, low-yield fusion explosions. More precisely, a small pellet of fuel is ejected from the vehicle and heated to ignition temperatures by high intensity laser beams. In the resulting nuclear explosion a quantity of propellant is heated by the released energy and expands as a high energy plasma, transferring momentum to the vehicle. A pusher plate transfers this impulse to a momentum conditioning unit which smooths the momentum transfer between pulses. The pusher plate is shielded by a magnetic field which, when compressed by the expanding plasma, increases in flux density and reverses the direction of the plasma, accelerating it away. The pusher plate also acts as a radiation shield for the vehicle.

NPP performance depends on several variables, including pulse mass, pulse material, and explosion rate. Some typical performance figures place specific impulses in the range of 4,000-10,000 sec, with thrust approaching 10^6 N. For missions requiring a high ΔV, NPP results in less propellant mass than chemical, solid-core nuclear, or gas-core nuclear propulsion systems.

There are some concerns about the feasibility of a NPP system. Although technologically feasible, development and test costs would be high. In addition, the 1963 Nuclear Test Ban Treaty places strict limits on atmospheric testing of nuclear explosions, which would impede the development of NPP. (11)

A.4.4 Nuclear Thermal Propulsion

The basic concept of the nuclear thermal rocket is a nuclear fission reactor core that transfers heat to an overflowing liquid propellant and increases its kinetic energy. Fission fragments heat a pressurized propellant, such as hydrogen, and change it into a superheated gas. The high energy propellant gas then expands at high velocity out of an exit nozzle. Since heat is transferred from the reactor components to the propellant fluid, the kinetic energy of the hydrogen gas is limited by the melting temperature of the reactor core. (12)

Gas core thermal rockets have great potential for high specific impulse and thrust because the reactor core is made of a fissionable cloud of material. The high energy, heavy fissionable particles are retained within the reactor after the light hydrogen particles collide with them. One possible way to accomplish this is for the liquid hydrogen propellant to enter the gaseous reactor chamber in a swirling motion. Thus, a strong vortex is generated within the chamber so that the heavier, less volatile fission cloud remains separate from the expanding light-weight hydrogen gas due to radial inertial forces. (13)

Nuclear thermal rockets provide both high thrust and specific impulse–a rare and ideal combination. Performance and power characteristics of nuclear thermal rockets are shown in Table A.4.

A.4.5 Antimatter Drives

Recent solid core thermal reactor concepts propose to use antiproton annihilation products instead of fission-heated fragments to heat a fluid propellant. The concept uses a circular tungsten shell to contain energetic pions as they heat the working fluid in the core. A magnetic field surrounds the tungsten shell to confine the pions from the reactor walls, as they spiral around the inner core. If the pions are allowed to react with the tungsten shell, dangerous neutrons are created that can melt the walls. Throughout the heating process liquid hydrogen flows between the magnetic field and the reactor shell to keep it from melting. Performance and power characteristics of antimatter drives are shown in Table A.4. (12)

Table A.4. Performance Characteristics of Some Advanced Propulsion Systems (6), (12), (13)

Performance Characteristics	Chemical Rocket	Nuclear Thermal Rocket	Antimatter Drive
Specific Impulse (s)	400 - 500	800-1000 Solid core 2000-6000 Gas core	1000 - 1500
Thrust	High	High	High
Ratio of Propellant Mass to Total Payload Mass in LEO	.8875	.6973	.6538
Efficiency	Low	Moderate	High

A.4.6 Propulsion System Evaluation

For Scenario I, Asteroid Retrieval, it is most feasible to accelerate a suitable asteroid using electric propulsion systems. Large arrays can be placed on an asteroid to give a ten to fifteen year flight time. Their high specific impulses allow them to move large payloads efficiently for long durations. Mass drivers and ion rockets have the best efficiency characteristics, especially at high specific impulses. Mass drivers, however, are probably too large and massive for serious consideration. Magnetoplasmadynamic (MPD) and Pulsed Inductive Thrusters (PIT) are not efficient enough for long duration missions. However, current research shows that PIT's have great potential for high thrust. This attribute, combined with their long service life may be useful for asteroid retrieval.

Presently human beings don't have enough experience in space or the necessary life support systems for long duration flights. Therefore, manned missions to asteroids and their subsequent return to Earth require high thrust propulsion systems. Chemical, nuclear, and antimatter rockets have high thrust characteristics, however, nuclear (pulsed and thermal) and antimatter systems are more fuel efficient with higher specific impulses. The development of technology for nuclear systems is in its advanced stages; however, antimatter systems are still purely conceptual. Light-weight antiproton-storage containers need to be developed before antiprotons can be used as efficient heat sources. Also, synthesis of antiprotons is presently very expensive. Thus, the most feasible high thrust, high specific impulse systems to develop are the nuclear thermal solid core and gas core rockets.

Table A.5. Comparison of Propulsion Systems for Asteroid Missions

Feasibility Criteria	Chemical Rockets	Electrostatic/ Electromagnetic	Nuclear Propulsion	Antiproton Drive
Propellant Mass	-	+	o	+
Cost	o	+	o	-
Overall Performance	o	o	+	+
Present Technology	+	+	o	-

Table A.6. Conceptual Soundness of Propulsion Systems Per Mission Scenarios

Mission Scenarios	Chemical Rockets	Electrostatic/ Electromagnetic	Nuclear Propulsion	Antiproton Drive
Asteroid Retrieval	-	+	o	o
Asteroid Mining / Refueling Waystation	-	+	o	o

(+)=good, (o)=neutral, (-)=bad

A.5 References

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A.6 FORTRAN program SPIRAL.FOR

This program is a simple model of a spiral trajectory from a given orbit to the Earth's orbit around the Sun. The program accepts the semi-major axis and eccentricity of the given asteroid. This information is first used to calculate the perihelion velocity of the asteroid in the original orbit. A distance increment is then created which divides the distance between the asteroid's orbit and Earth's orbit into several evenly-spaced steps. A series of Hohmann transfers is then used to model a spiral, with the aphelion of the present Hohmann ellipse being reduced by the specified increment after each iteration. The asteroid's orbit is reduced in size each time until Earth's orbit is reached. After all iterations have been completed, a DV for capture at Earth is then calculated and added to the total DV. The program neglects plane change and assumes that the asteroid is targeted correctly to encounter the Earth in a flyby.

```
      PROGRAM SPIRAL
c     This program models a spiral trajectory from solar orbit
c     to Earth orbit
c
c     Written by Scott Patano for ATS
c
c Define variables
      REAL MU,A(10),E(10),I,RP(10),VP(10),DELV(11),TIME(10),
     +     INCR,RA(10),VA(10),TDELV,TTIME,MUE,RPH,VC,VE,
     +     VINF,AH,VPH,DIAM,AA,NUM
      INTEGER J,K,Z
c Input orbital elements
      OPEN(6,FILE='SPIRAL.OUT')
      OPEN(7,FILE='ASTEROID.NEW')
      DO 10 Z=1,2958
      READ(7,*)NUM,A(0),E(0),I,DIAM
      AA=A(0)
      A(0)=A(0)*1.4959787E11
c Calculate needed parameters from elements
      MU=1.3271244E20
      RP(0)=A(0)*(1-E(0))
      VP(0)=SQRT((MU*(1+E(0)))/(A(0)*(1-E(0))))
      INCR=(RP(0)-1.4959787E11)/9
c Calculate necessary plane change
C Brian sets plane change to 0
      I=I*(3.1415926)/180.
      DELV(0)=2*VP(0)*SIN(I/2)
      DELV(0)=DELV(0)/1000.
      DELV(0)=0
c Calculate delta-vees, time of flights for 8 Hohmann transfers
      DO 1 J=1,9
      RA(J)=RP(J-1)
      RP(J)=RP(J-1)-INCR
      E(J)=(RA(J)-RP(J))/(RA(J)+RP(J))
      A(J)=(RA(J)+RP(J))/2
      VP(J)=SQRT((MU*(1+E(J)))/(A(J)*(1-E(J))))
      VA(J)=VP(J)*((1-E(J))/(1+E(J)))
      DELV(J)=VP(J-1)-VA(J)
      TIME(J)=(3.1415926)*SQRT(((RA(J)+RP(J))**3)/(8*MU))
 1    CONTINUE
c Calculate delta vee for insertion into Earth orbit
      MUE=3.9860044E14
      RPH=1.922E8
      VC=1440.1
```

```
      VE=29784.7
      VINF=VP(9)-VE
      AH=MUE/(VINF**2)
      VPH=SQRT(MUE)*SQRT((2/RPH)+(1/AH))
      DELV(10)=VPH-VC
      DELV(10)=DELV(10)/1000.
c Output delta vees, times of flight
      TDELV=DELV(0)+DELV(10)
      TTIME=0.
      DO 2 K=1,9
      DELV(K)=DELV(K)/1000.
      TIME(K)=TIME(K)/(86400.*365.25)
      TDELV=TDELV+ABS(DELV(K))
      TTIME=TTIME+TIME(K)
 2    CONTINUE
      A(0)=A(0)/1.4959787E11
      I=I*(180/3.1415926)
      WRITE(6,15)NUM,DELV(0),DELV(10),TDELV,TTIME,DIAM,AA,
     +          E(0),I
 10   CONTINUE
 15   FORMAT(2X,F5.0,2X,F8.5,2X,F8.5,2X,F8.3,3X,F5.2,3X,F5.1,2X,
     +       F5.3,2X,F5.3,2X,F6.3)
      END
```

Appendix B: FORTRAN Optimization Code

The FORTRAN program ASTOUT.FOR was obtained and modified to determine the minimum ΔV required to place the spacecraft on a rendezvous trajectory to a given asteroid. The inputs required by the program are the orbital elements of the asteroids being considered. The program has three loops. The outer loop iterates through the 4800+ asteroid file obtained from JPL (epoch 1/1/1950). The next loop of the program iterates between the years 1997 and 2010 in user defined steps while the inner loop iterates the time of flight (TOF) between user defined minimum and maximums in user defined steps. The inner loop calls a SLMBRT Lambert targeting subroutine which calculates the ΔV for each specific launch date and TOF. The program writes the minimum ΔV with its corresponding launch date and time of flight for each asteroid to an output file.

This program was later modified to provide launch windows by reducing the range of the launch date and TOF chosen for their corresponding low ΔV. Finally it was modified to provide the radius vectors and angles of the Earth, asteroid and Hawking spacecraft during the flight to asteroid 3677.

The subroutines called in the program such as SLMBRT and OETORC are readily available from the University of Texas at Austin, Department of Aerospace Engineering or from any NASA mission planning software library.

```
C
C     ************************************************
C     ** THIS PROGRAM WAS ORGINALLY WRITTEN AS A COMET    *
C     ** RENDEVOUS PROGRAM
C     ** TONY ECONOMOPOULOS                               *
C     ** SPRING 1991                                      *
C     ** PROGRAM VERSION: ALPHA                           *
C     ************************************************
C     ***************************************************************
C
C         THIS PROGRAM HAS BEEN MODIFIED BY MEDHA DATE, SCOTT PATANO AND
C         BRIAN RADOVICH TO READ THE ORBITAL ELEMENTS OF AN ASTEROID FROM
C         FROM AN INPUT FILE AND CALCULATE THE SMALLEST DELTA V FROM THE
C         EARTH TO THE ASTEROID.
C
C         THE PROGRAM INPUT IS A FILE WITH THE ASTEROID ORBITAL
C         ELEMENTS. THE PROGRAM OUTPUT IS A FILE WITH THE SMALLEST
C         DELTA V WITH THE CORRESPONDING TIME OF FLIGHT FOR EAC H LAUNCH DATE.
C         THE ORBITAL ELEMENTS IN THE INPUT FILE ARE READ AS FOLLOWS
C
C                  OECO(1)= SEMI-MAJOR AXIS
C                  OECO(2)= ECCENTRICITY
C                  OECO(3)= INCLINATION
C                  OECO(4)= LONGITUDE OF THE ASCENDING NODE
C                  OECO(5)= ARGUMENT OF PERISAPSIS
C                  OECO(6)= TIME OF LAST PERIAPSIS PASSAGE
C                           SINCE USER CHOSEN EPOCH
C                  OECO(7)= MU OR THE GRAVITATIONAL PARAMETER
C                  OECO(8)= TIME SINCE USER CHOSEN EPOCH
C
C     ***************************************************************
C        THIS PROGRAM CALLS SUBROUTINES:
C                       RCE, DIF, SLMBRT, AND OETORC
```

```
C
C     ***********************************************************************
C
      PROGRAM ASTOUT
C     ***********************************************************
C     Declare Variables:  RC=Asteroid Coordinates
C
      IMPLICIT DOUBLE PRECISION (A-H,O-Z)
      DOUBLE PRECISION OECO(8),RC(3),RE(3),VC(3),VE(3),
     +                 OQ(8),VTE(3),VTC(3),DITI,R(50),S(50)
      DOUBLE PRECISION AU2KM,KMU,INC,BUFFER(21)
      INTEGER I,NUM,RTOF,riti
      CHARACTER*18 NAME
      CHARACTER*6 IREF,FAMILY,PHOTO
      DOUBLE PRECISION SEM,EC,INC,RAD,DAY,PER,RP,RA,SYNP,OM,W,PERI
C
      OPEN(9,FILE='pmeters2')
C      OPEN(15,FILE='rad.front')
C      OPEN(10,FILE='velocity')
      OPEN(19,FILE='dv.all')
      OPEN(UNIT=10,file='astcom.fmt',form='formatted',status='old')
C
C     ** CONSTANTS AND CONVERSION FACTORS
C
      AU2NMI=0.8077640010799D8
      AU2KM=1.4959965D8
      KMU=1.3271544D11
      SD2SEC=86636.55536D0
      D2R=DACOS(-1.0D0)/180.0D0
      SMU=0.2089242635906454D11
      T0D=99.0D0
C
C  ********** STEP IS STEP SIZE FROM LAUNCH DATE TO LAUNCH DATE ********
C
      STEP=15.0D0
C
C     **************************************************************
C         OPENING FILE AND READING ORBITAL ELEMENTS OF ASTEROID
C     **************************************************************
C
      DO 10 I=1,5000
         READ(10,100,END=99)NUM,NAME,IREF,FAMILY,PHOTO,BUFFER
 100     FORMAT(I4,A18,3A6,5X,2E15.9/(5E15.9))
C         write(*,*)num
         SEM=BUFFER(1)
         EC=BUFFER(2)
         INC=BUFFER(3)
         OM=BUFFER(4)
         W=BUFFER(5)
         RMEANAN=BUFFER(6)
         REPOCH=BUFFER(7)
         RRADIUS=BUFFER(8)
         RGM=BUFFER(9)
         RDEC=BUFFER(10)
         RRASC=BUFFER(11)
         RDAY=BUFFER(12)
         PERI=BUFFER(13)
         RPERIOD=BUFFER(14)
```

```
      RPYR=BUFFER(15)
      RRP=BUFFER(16)
      RRA=BUFFER(17)
C
C
C
      OECO(1)=SEM*AU2NMI
      OECO(2)=EC
      OECO(3)=INC*DACOS(-1.0D0)/180.0D0
      OECO(4)=OM*DACOS(-1.0D0)/180.0D0
      OECO(5)=W*DACOS(-1.0D0)/180.0D0
      OECO(6)=PERI*SD2SEC
      OECO(7)=SMU
C
C  *******************************************************************
C  ****TID IS BEGINNING LAUNCH DATE AND TOD IS ENDING LAUNCH DATE*****
C  ****GIVEN IN NUMBER OF DAYS SINCE 1950                         *****
C
      TID=17000.0D0
      TFD=22000.0D0
      DVPR=99.99D69
C
C
C     ****************LOOP FOR TIMES OF FLIGHT*************************
C
      DO 399 RTOF=1,30
         TOFD = 150.D0 + (10.D0*(DBLE(RTOF)-1.D0))
C
C    *****************************************************************
C    ***SETTING OQ 1 AND 7  FOR CALLING SLMBRT (TIME OF FLIGHT AND MU)***
C
         OQ(1)=TOFD*SD2SEC
         OQ(7)=SMU
C
         T0S=T0D*SD2SEC
C
C    ***************LOOP FOR LAUNCH DATES******************************
C
         DO 200 ITI=IDINT(TID),IDINT(TFD),IDINT(STEP)
C
C    ***OECO(8), TIME SINCE USER CHOSEN EPOCH GOES TO OETORC ***********
C
            OECO(8)=ITI*SD2SEC+TOFD*SD2SEC
C
C    *******T IS LAUNCH DATE IN SECONDS FOR RCE********************
C
            T=ITI*SD2SEC
C
C    ****** CALLING SUBROUTINES OETORC AND RCE *******
C
            CALL OETORC (OECO,RC,VC)
            CALL RCE (T,T0S,RE,VE)
            CALL DIF (RE,VE,RC,VC,RDIF,VDIF)
            ARC1=RC(1)
            ARC2=RC(2)
            ARC3=RC(3)
            RC(1)=RC(1)+(VC(1)/VDIF)*50.D0
            RC(2)=RC(2)+(VC(2)/VDIF)*50.D0
```

```
              RC(3)=RC(3)+(VC(3)/VDIF)*50.D0
              ARC=DSQRT(RC(1)**2+RC(2)**2+RC(3)**2)
C     AVC=DSQRT(2.*(VDIF**2/2.-SMU/RDIF+SMU/ARC))
c     AVC1=AVC*RC(1)/ARC
c     AVC2=AVC*RC(2)/ARC
c     AVC3=AVC*RC(3)/ARC
C
              RADCO=DSQRT(RC(1)**2+RC(2)**2+RC(3)**2)
              VELCO=DSQRT(VC(1)**2+VC(2)**2+VC(3)**2)
C
C
C     ********** CALLING SUBROUTINE SLMBRT ******************
C
              CALL SLMBRT (OQ,RE,RC,VTE,VTC,.FALSE.,IERR)
              IF (IERR.EQ.1.0) THEN
                 WRITE (6,*) 'ERROR IN SLMBRT'
              ENDIF
C
C   ******* CALLING DIF TO CALCULATE DV AT EARTH AND ASTEROID ************
C
              CALL DIF (VE,VC,VTE,VTC,DVE,DVC)
              ARC1=(ARC1-RC(1))*AU2KM/AU2NMI
              ARC2=(ARC2-RC(2))*AU2KM/AU2NMI
              ARC3=(ARC3-RC(3))*AU2KM/AU2NMI
              AVC1=(AVC1-VC(1))/.53960D0
              AVC2=(AVC2-VC(2))/.53960D0
              AVC3=(AVC3-VC(3))/.53960D0
              DVTOT=(DVE+DVC)/.53960D0
              RDVE=DVE/.53960D0
              RDVC=DVC/.53960D0
C
C ************** FINDING LOWEST DELTA V ****************
C
              if (dvtot.le.DVPR) then
                 rrriti=dble(iti)
                 rrrdvtot=dvtot
                 DVPR = DVTOT
                 adate=iti+tofd
C     write(19,431)num,iti,tofd,adate,rrrdvtot,RDVE,RDVC
C     write(15,432)ARC1,ARC2,ARC3
C     write(10,*)VC(1),VC(2),VC(3)
C     write(*,*)num,rrriti,tofd,rrrdvtot,RC(1),RC(2),RC(3)
              endif
 200        CONTINUE
 399      CONTINUE
          IF (DVPR.LE.8.0d0) THEN
             WRITE(19,*)NUM,DVPR
          ENDIF
 10   CONTINUE
C
 111  FORMAT(1X,I5,3X,20(F5.1,1X))
 112  FORMAT(1X,I5,3X,20(F7.1,1X))
 113  FORMAT(1X,I5,5X,F15.9,5X,F7.1,5X,F15.9)
 114  FORMAT(1X,I8,3(5X,F15.9))
 431  format(i5,3x,i8,3x,f5,3x,f8,3x,3(f8.2),1x,f13.2)
 432  format(2x,3(2x,f13.2))
C
      CLOSE(2)
```

```
      CLOSE(3)
      CLOSE(4)
      CLOSE(7)
 99   STOP
      END
C
C*****************************************************************
C
C     ** THIS IS SUBROUTINE RCE **
C     ** IT CALCULATES THE RECTANGULAR COORDINATES
C     ** OF EARTH USING A SIMPLIFIED MODEL (CIRCULAR
C     ** ORBIT AROUND THE SUN WITHOUT PERTURBATIONS)
C
      SUBROUTINE RCE (T,T0,RE,VE)
      IMPLICIT DOUBLE PRECISION (A-H,O-Z)
      DOUBLE PRECISION RE(3),VE(3)
C
C ***********EARTH ROTATION CONSTANTS************************
      RSE=.8077640010799D8
      PER=(365.25D0)*(86636.55536D0)
      OMEGA=2*DACOS(-1.0D0)/PER
C
C ******************CALCULATION OF TIME***************************
C **** TO IS THE NUMBER OF SECONDS FROM VERNAL EQUINOX TO JAN.1******
C
C     TN = DATE OF LAUNCH SINCE 1/1/1950
C     IORBT = NUMBER OF YEARS SINCE 1950
C     TRT = THE NUMBER OF SECONDS SINCE JAN 1
C
      TN=T
      IORBT=IDINT(TN/PER)
      TRT=TN-IORBT*PER
C
C ************CALCULATION OF EARTH'S POSITION**************
C
      RE(1)=RSE*DCOS(OMEGA*(TRT+T0))
      RE(2)=RSE*DSIN(OMEGA*(TRT+T0))
      RE(3)=0.0D0
C
C **********CALCULATION OF EARTH'S VELOCITY************
C
      VE(1)=-OMEGA*RSE*DSIN(OMEGA*(TRT+T0))
      VE(2)=OMEGA*RSE*DCOS(OMEGA*(TRT+T0))
      VE(3)=0.0D0
C
      RETURN
      END
C
C*****************************************************************
C
C     ** THIS SUBROUTINE (DIF) CALCULATES THE DISTANCE AND
C     ** VELOCITY OF THE COMET WITH RESPECT TO EARTH
C     ** IT ALSO CALCULATES THE DVs FROM THE RESULTS
C     ** OF THE LAMBERT TARGETING ROUTINE
C
      SUBROUTINE   DIF (RE,VE,RC,VC,RADCE,VELCE)
      IMPLICIT DOUBLE PRECISION (A-H,O-Z)
      DOUBLE PRECISION RE(3),VE(3),RC(3),VC(3),RADCE,VELCE
```

```
C
      SUMR=0.0D0
      SUMV=0.0D0
      DO 100 I=1,3
        SUMR=SUMR+(RC(I)-RE(I))**2
        SUMV=SUMV+(VC(I)-VE(I))**2
 100  CONTINUE
      RADCE=DSQRT(SUMR)
      VELCE=DSQRT(SUMV)
 111  FORMAT(1X,F6.1,3X,6(F4.1,3X))
C
      RETURN
      END
```

Clohessy-Willshire (CW) Targeting

The CW equations are used to model small orbital maneuvers between two satellites orbiting the same body, with one of these satellites used as an orbiting reference point. The variables in the equations include the time of flight to complete the maneuver, the initial and final X, Y, Z positions, and the initial and final X, Y, Z velocities in the orbiting reference frame. The CW model assumes a circular orbit for the reference satellite and that the maneuvers are only affected by the body being orbited.

The CW equations were used in obtaining a rough approximation of the ΔV's required of the lander/SRC while docking with the asteroid. The TK Solver model uses the asteroid as the orbiting reference point and the Sun as the body being orbited. The inputs to the model were the initial and final X, Y, Z positions, with the time of flight given as a constant. The initial X, Y, Z position is the point at which the SRC separates from the orbiter section of the Hawking spacecraft. The final position is at the asteroid at X=0, Y=0, and Z=0. The outputs of the model were the initial and final velocities. The ΔV may be calculated from this data by subtracting the velocity of the SRC upon orbiter separation from the initial velocity required to reach the asteroid. The TK Solver model was iterated through 43 sets of input.

The data obtained from this model are only rough estimates because of the approximations made. First, the orbit of the asteroid was assumed circular. This is not accurate because the orbit actually has an eccentricity of 0.36. Second, the asteroid is massive enough to affect the trajectory of the spacecraft at close distances. Because of these assumptions, any ΔV's calculated from this model should be doubled to provide a factor of safety when considering thruster and ΔV requirements for the SRC.

Appendix C: ROCKET2 Program

Program ROCKET2 was written to obtain a detailed breakdown of the mass requirements of the Hawking Explorer mission, based on a specified set of input parameters. The results of the program provide a detailed description of Hawking's component vehicles and the mission scenario.

Input Parameters for NROCKET2.INP input data file:

Mass (kg):	SRC, Lander, Orbiter
Delta-V's (m/s):	upper stage 1, upper stage 2, asteroid insertion, asteroid docking, Earth transfer, elliptical Earth orbit, circularize into LEO
Isp's (seconds):	upper stage 1, upper stage 2, hawking cruise, hawking docking, hawking transfer to Earth
Thrust (N):	upper stage 1, upper stage 2, hawking mains, SRC main
Upper stage type:	upper stage 1, upper stage 2 (liquid=1, solid=2, none=0)
Mass fraction:	upper stage 1, upper stage 2
Mass fraction:	Lander, SRC
Mass of Sample (kg):	####
Tank pressure (MPa):	fuel tanks, oxidizer tanks, pressurant tanks
Pressurant param.:	ratio of specific heats, gas constant (SI), temperature (K)
Upper stage 3 ?:	upper stage 3 (liquid=1, solid=2, none=0)
Mass fraction:	upper stage 3

Example of NROCKET2.INP input data file:

```
50.,238.7,902.4
4000.,1600.,1700.,200.,4500.,1000.,3000.
446.,292.,445.,445.,445.
147000.,40000.,1250.,500.
1.,2.
.84,.93
.88,.93
3.0
13.79E+6,13.79E+6,110.32E+6
1.2,2078.4419,300.
1.
.88
```

Equations

Begin with equation (1)

$$\Delta V = I_{sp} g \ln \frac{M_o}{M_f} \tag{1}$$

$$\frac{\Delta V}{I_{sp} g} = \ln \frac{M_o}{M_f}$$

and isolate the mass ratio term to get

$$e^{\frac{\Delta V}{I_{sp}g}} = \frac{M_o}{M_f} \tag{2}$$

Equations (3) through (6) represent the ratios of initial mass to final mass for each burn the SRC performs with its onboard propulsion system. Equation (3) represents the main transfer burn from the asteroid to Earth. The NROCKET2 program allows the user to use either a separate stage or the SRC's onboard propulsion system for this burn. If a jettisonable stage is used, then equation (3) is not used in the overall calculation of the SRC's propulsion system.
Now then, equation (7) relates the SRC's total fuel mass to its propellant tank mass through the propellant mass fraction parameter, M_{FR}. The SRC's propellant tank mass, $M_{TNK_{SRC}}$, is represented by equation (8).

Equation (8) is arrived at by first substituting the right-hand-side of (7) for the fuel variables in the numerator of (3). Then a series of substitutions--(6) into (5), (5) into (4), and (4) into (3)--are made by multiplying the first equation by the denominator of its right-hand side and substituting the left-hand-side for the denominator of the right-hand-side of the second equation. Finally, equation (8) is solved in terms of all the delta-v's, the assumed mass fraction, the specific impulse, Earth's gravitational constant, and the payload mass--the SRC.

$$e^{\frac{\Delta V_{AE}}{I_{sp}g}} = \frac{M_{SRC} + M_{TNK_{SRC}} + M_{F_{AE}} + M_{F_{MC}} + M_{F_{EI}} + M_{F_{LEO}}}{M_{SRC} + M_{TNK_{SRC}} + M_{F_{MC}} + M_{F_{EI}} + M_{F_{LEO}}} \tag{3}$$

$$e^{\frac{\Delta V_{MC}}{I_{sp}g}} = \frac{M_{SRC} + M_{TNK_{SRC}} + M_{F_{MC}} + M_{F_{EI}} + M_{F_{LEO}}}{M_{SRC} + M_{TNK_{SRC}} + M_{F_{EI}} + M_{F_{LEO}}} \tag{4}$$

$$e^{\frac{\Delta V_{EI}}{I_{sp}g}} = \frac{M_{SRC} + M_{TNK_{SRC}} + M_{F_{EI}} + M_{F_{LEO}}}{M_{SRC} + M_{TNK_{SRC}} + M_{F_{LEO}}} \tag{5}$$

$$e^{\frac{\Delta V_{LEO}}{I_{sp}g}} = \frac{M_{SRC} + M_{TNK_{SRC}} + M_{F_{LEO}}}{M_{SRC} + M_{TNK_{SRC}}} \tag{6}$$

$$M_{F_{AE}} + M_{F_{MC}} + M_{F_{EI}} + M_{F_{LEO}} = \frac{M_{TNK_{SRC}}}{1 - M_{FR}} - M_{TNK_{SRC}} \tag{7}$$

$$M_{TNK_{SRC}} = \frac{\left(e^{\frac{\Delta V_{AE}+\Delta V_{MC}+\Delta V_{EI}+\Delta V_{LEO}}{I_{sp}g}}\right)}{\frac{1}{1 - M_{FR}} - e^{\frac{\Delta V_{AE}+\Delta V_{MC}+\Delta V_{EI}+\Delta V_{LEO}}{I_{sp}g}}} M_{SRC} \tag{8}$$

By substituting (8) into (6), (5), (4), and (3), respectively, the fuel mass for each burn is calculated. The NROCKET2 program back-calculates from the end of the mission to the beginning, in order to solve for the initial mass of the vehicle with all of its stages. After the total mass of the SRC (structure, payload, and fuel) is obtained, it is used as the payload mass for calculating the fuel requirement of the Lander. The total mass of the Lander, SRC, and

Orbiter are then used to calculate the fuel required by the Earth-to-asteroid upper stage(s). The equations are similar to those above, except the variable names are different. Mass fractions are assumed for the Lander and the upper stages as well as the SRC.

NROCKET2 is a powerful tool in designing the overall mission. If a particular parameter (i.e. delta-v, mass fraction, thrust, Isp, component vehicle mass, removing/adding a stage, etc.) is changed, the effect of that change on the propulsion requirements can be determined immediately. Thus, the mission's design can be iterated until the mission criteria are satisfied. Variable definitions for Equations (1)-(8):

ΔV:

- $_{AE}$ = Asteroid-to-Earth transfer
- $_{MC}$ = Midcourse Correction
- $_{EI}$ = Earth Insertion (highly elliptical orbit)
- $_{LEO}$ = Circularization burn into LEO

I_{sp} & M:

- $_{o}$ = Initial mass before burn
- $_{f}$ = Final mass after burn
- $_{SRC}$ = Sample Return Craft
- $_{TNKSRC}$ = Tank mass of SRC
- $_{FAE}$ = Fuel mass required for asteroid-to-Earth transfer
- $_{FMC}$ = Fuel mass required for midcourse correction
- $_{FEI}$ = Fuel mass required for Earth insertion
- $_{FLEO}$ = Fuel mass required for circularization into LEO
- $_{FR}$ = Assumed mass fraction

NROCKET2.DAT (results):

UPPER STAGES, 1=LIQUID 2=SOLID 0=NONE #1= 1. #2= 2. #3= 1.
HAWKING DRY MASSES: SRC = 50.0 LANDER = 238.7 ORBITER = 902.4

PROPULSION SUMMARY STARTING FROM LEO

PHASE	DELTA V (m/s)	ISP (s)	BURN TIME (s)	FUEL MASS (kg)	FUEL FRAC.	TOTAL IMPULSE (N-s)	MASS RATIO	HAWKING MASS (kg)
1	4,000.	446.	384.	12,904.	.8400	56,460,268.	----	21,537.
2	1,600.	292.	189.	2,643.	.9300	7,569,518.	.2867	6,175.
MCB	500.	445.	1,260.	361.	.8800	1,574,681.	.5398	3,333.
3	1,700.	445.	3,348.	959.	.6482	4,185,447.	.8918	2,972.
4	200.	445.	4,345.	50.	.0320	217,242.	.3739	1,111.
5	4,500.	445.	1,436.	411.	.8800	1,794,745.	.5751	639.
MCB	500.	445.	162.	19.	.9300	81,217.	.2690	172.
6	1,000.	445.	274.	31.	.7735	137,017.	.8918	153.
7	3,000.	445.	529.	61.	.5096	264,540.	.7953	122.
TOTAL	17,000.			17,438.		72,284,664.	.0057	

SRC MASS (kg):			
BEFORE MAIN TRANSFER BURN =	639.10089111		
BEFORE EARTH RENDEZVOUS =	153.30912781		
BEFORE EARTH INSERTION =	121.92240906		
SRC FUEL TANK MASS =	8.32395172		
LANDER MASS (kg):			
BEFORE ASTEROID INSERTION =	1433.94555664		
BEFORE SURFACE RENDEZVOUS =	475.17907715		
AFTER SURFACE RENDEZVOUS =	425.41522217		
LANDER FUEL TANK MASS =	186.71520996		
MASS OF UPPER STAGES (kg):	PROPELLANT	TANKS	
LEO TO ASTEROID, #1 =	12904.44	2457.99	
LEO TO ASTEROID, #2 =	2642.51	198.90	
ASTEROID-> EARTH #3 =	411.12	56.06	

PROPELLANT/PRESSURANT MASS (kg) & VOLUME (m^3)

PROPELLANT	LANDER		UPPR.STG. #3		SRC	
	V	M	V	M	V	M
FUEL	0.6606	288.7	0.1984	86.7	0.0534	23.3
OXYDIZER	0.7554	1149.0	0.2268	345.0	0.0610	92.8
PRESSURANT	0.2676	45.1	0.0804	13.5	0.0216	3.6

PROPELLANT & PRESSURANT TANK MASSES (kg) & DIMENSIONS (m)

	LANDER			UPPR.STG. #3			SRC		
# OF TANKS	R	T	M	R	T	M	R	T	M
1 FUEL	0.54	.1086E-01	182.75	0.36	.7273E-02	54.87	0.23	.4695E-02	14.76
TNK CLAD		.7603E-02			.5091E-02	18.26		.3286E-02	4.91
E.T.		.1125E-01			.7532E-02			.4862E-02	
1 OXYD	0.56	.1136E-01	208.97	0.38	.7605E-02	62.75	0.24	.4909E-02	16.88
TNK CLAD		.7950E-02	69.53		.5324E-02	20.88		.3437E-02	5.62
E.T.		.1176E-01			.7876E-02			.5084E-02	
TOTALS			522.07			156.75			42.17
FOPTNK	0.40	.6340E-01	558.03	0.27	.4245E-01	167.55	0.17	.2740E-01	45.07
F PTNK	0.31	.4917E-01	260.34	0.21	.3293E-01	78.17	0.13	.2125E-01	21.03
O PTNK	0.32	.5142E-01	297.69	0.22	.3443E-01	89.38	0.14	.2223E-01	24.04
TOTALS			1080.09			324.31			87.24

SOME DEFINITIONS REGARDING ROCKET2 PROGRAM OUTPUT:

HAWKING MASS; mass of the spacecraft before the particular burn is performed.
V; volume
M; mass
R; radius
T; thickness
1 OXYD; one oxidizer tank.
1 FUEL; one fuel tank.
E.T.; combined effective thickness of cladding and tank walls used for stress calculations.
FOPTNK; all pressurant for both fuel and oxidizer in one tank.
FPTNK; single pressurant tank for fuel.
OPTNK; single pressurant tank for oxidizer.

```
      PROGRAM ROCKET2
C
C     WRITTEN BY NEIL ERIAN
C     FOR ASE274L/174M
C
C     PROJECT S.T.O.N.E.R.- Systematic Transfer of Near-Earth Resources
C
C     THIS PROGRAM THOROUGHLY DEVELOPS THE EQUATION:
C
C                                DV=ISP*g*ln(Mo/Mf)
C
C     FOR AN ASTEROID EXPLORER SAMPLE RETURN SPACECRAFT. PROPELLANT AND
C     PRESSURANT TANK SIZES ARE CALCULATED AS WELL AS SPACECRAFT COMPONENT
C     MASSES.
C
C     VARIABLES:  DV    CHANGE IN VELOCITY FOR PARTICULAR MISSION PHASE
C                 MFDV  FUEL NECESSARY TO PERFORM DELTA-V (DV)
C                 ISP   ISP OF ENGINE FOR PARTICULAR BURN
C                 MSRC  DRY MASS OF SAMPLE RETURN CRAFT
C                 MORB  DRY MASS OF ORBITER
C                 ML    DRY MASS OF LANDER (I.E. LANDING PACKAGE)
C               TOTISP  TOTAL IMPULSE FOR THE ENTIRE MISSION
C                 TUS1  TYPE OF UPPER STAGE #1 (LEO TO ASTEROID TRAJECTORY)
C                 TUS2  TYPE OF UPPER STAGE #2 (LEO TO ASTEROID TRAJECTORY)
C                 TUS3  TYPE OF UPPER STAGE #2 (ASTEROID TO EARTH TRAJECT.)
C                 ITOT  TOTAL IMPULSE FOR PARTICULAR BURN
C                MCOMP  FINAL MASS OF THE SPACECRAFT AFTER PARTICULAR BURN
C                  MFR  PROPELLANT MASS FRACTION
C                TPROP  TOTAL PROPELLANT USED DURING ENTIRE MISSION
C                TOTDV  TOTAL DELTA-V REQUIRED FOR ENTIRE MISSION
C                    G  EARTH GRAVITATIONAL CONSTANT
C               TNKSTR  MASS OF UPPER STAGE STRUCTURE
C               TNKSRC  MASS OF SRC FUEL TANKS
C               TNKLAN  MASS OF LANDER TANKS
C               TOTSRC  TOTAL MASS OF SRC (TANKS,PROPELLANT,STRUCTURE)
C               TOTLAN  TOTAL MASS OF LANDER (TANKS,PROPELLANT,STRUCTURE)
C                 HAWK  MASS OF HAWKING AT ANY PART OF MISSION
C
C                   TO  PRESSURANT TANK DESIGN TEMPERATURE
C               RPRESS  GAS CONSTANT OF PRESSURANT
C               KPRESS  RATIO OF SPECIFIC HEATS FOR PRESSURANT GAS
C                   VF  FUEL VOLUME
```

```
C                   VO  OXYDIZER VOLUME
C               RHOH2O  DENSITY OF WATER
C                    R  OXYDIZER/FUEL RATIO
C               SGFUEL  SPECIFIC GRAVITY OF FUEL
C               SGOXYD  SPECIFIC GRAVITY OF OXYDIZER
C                   MF  MASS OF FUEL
C                   MO  MASS OF OXYDIZER
C                  VPO  VOLUME OF PRESSURANT NEEDED FOR OXYDIZER
C                  MPO  MASS   OF     "         "      "      "
C                  VPF  VOLUME OF     "         "      "      "
C                  MPF  MASS   OF     "         "      "      "
C                  RPT  RADIUS FOR SINGLE PRESSURANT TANK
C                  MPT  MASS    OF  "           "       "
C                 THPT  WALL THICKNESS OF PRESSURANT TANK
C                  RFT  RADIUS OF FUEL TANK
C                  MFT  MASS OF FUEL TANK
C                 THFT  THICKNESS OF FUEL TANK
C                  ROT  RADIUS OF OXYDIZER TANK
C                  MOT  MASS OF OXYDIZER TANK
C                 THOT  THICKNESS OF OXYDIZER TANK
C
C               MCLDOT  MASS OF OXYDIZER TANK CLADDING
C               MCLDFT  MASS OF FUEL TANK CLADDING
C                 PTNK  PRESSURE OF PROPELLANT AND PRESSURANT TANKS
C               SIGCLD  CLADDING MATERIAL ALLOWABLE STRESS
C               RHOCLD  DENSITY OF CLADDING MATERIAL
C               RHOTNK  DENSITY OF TANK MATERIAL
C        EFFFTH,EFFOTH  COMBINED CLAD AND BACKING EFFECTIVE THICKNESS
C                       FOR OXYDIZER AND FUEL TANKS
C                THCLD  THICKNESS OF CLADDING
C
C
C     *************************
C     *                       *
C     *  DIMENSION VARIABLES  *
C     *                       *
C     *************************
C
      IMPLICIT CHARACTER*1 (A-Z)
      REAL DV(10),ISP(10),MSRC,MORB,TOTISP,TUS1,TUS2,ITOT(10),MFDV(11),
     +ML,MFR(10),TPROP,TOTDV,G,TOTSRC,TNKSRC,TOTLAN,TNKLAN,HAWK1,BT(10),
     +TNKSTR(10),A,B,MCOMP(10),MFRLAN,TUS3,MFRSRC,MSAMP,HAWK,TH(10)

      REAL TO,RPRESS,VF(3),VO(3),RHOH2O,R,SGFUEL,SGOXYD,MF(3),MO(3),
     +KPRESS,VPO(3),MPO(3),RPT(3),RFT(3),ROT(3),MPT(3),MFT(3),
     +MOT(3),THPT(3),THFT(3),THOT(3),VPF(3),MPF(3)

      REAL MCLDOT(3),MCLDFT(3),UTS,SIGY,PTNK(3),SIGCLD,RHOCLD,RHOTNK,
     +EFFOTH(3),EFFFTH(3),MNEWFT(3),MNEWOT(3),THCLDF(3),THCLDO(3),
     +MNEW(3),RFPT(3),ROPT(3),THFPT(3),THOPT(3),MFPT(3),MOPT(3)
      INTEGER K,J,H,L,N
C
C     ********************************
C     *                              *
C     * PREPARE INPUT/OUTPUT FILES   *
C     *       FOR DATA HANDLING      *
C     *                              *
C     ********************************
```

```
C
      OPEN(UNIT=3,FILE='NROCKET2.DAT')
      OPEN(UNIT=4,FILE='NROCKET2.INP')
C
C
C     *******************
C     *                 *
C     *   GET SYSTEM    *
C     *   PARAMETERS    *
C     *                 *
C     *******************
C
C      WRITE(*,*)'INPUT MASS OF:'
C      WRITE(*,*)'          SRC --WITHOUT ASTEROID REGOLITH SAMPLES'
C      WRITE(*,*)'       LANDER --WITHOUT PROPELLANT TANKS'
C      WRITE(*,*)'      ORBITER --ALL COMPONENTS                    '
      READ(4,*)MSRC,ML,MORB
C      WRITE(*,*)
C      WRITE(*,*)'                                                   PHASE'
C      WRITE(*,*)'DELTA-V: HAWKING, LEO TO ASTEROID UPPER STAGE #1  (1)'
C      WRITE(*,*)'         HAWKING, LEO TO ASTEROID UPPER STAGE #2  (2)'
C      WRITE(*,*)'         HAWKING INSERTION AT ASTEROID            (3)'
C      WRITE(*,*)'         SURFACE RENDEZVOUS BY SRC/LANDER         (4)'
C      WRITE(*,*)'         PRIMARY BOOST OF SRC TO EARTH TRAJECTORY (5)'
C      WRITE(*,*)'         SRC INSERTION INTO e=.9 EARTH ORBIT      (6)'
C      WRITE(*,*)'         CIRCULARIZATION BURN AT 300 KM ALTITUDE  (7)'
C      WRITE(*,*)
C      WRITE(*,*)
C      WRITE(*,*)'INPUT DELTA-V FOR EACH PHASE SEPARATED BY COMMAS'
C      WRITE(*,*)'NOTE: ENTER 0 FOR PHASE 2 IF NO SECOND'
C      WRITE(*,*)'UPPER STAGE.                       '
      READ(4,*)DV(2),DV(3),DV(5),DV(6),DV(7),DV(9),DV(10)
C
      DV(4)=500.
      DV(8)=500.
C
C      WRITE(*,*)
C      WRITE(*,*)'INPUT ISP FOR (1), (2), (3)/(4), (5), (6)/(7)'
C      WRITE(*,*)'NOTE: ENTER 0 FOR PHASE 2 AS NECESSARY.      '
      READ(4,*)ISP(2),ISP(3),ISP(5),ISP(7),ISP(9)
      ISP(6)=ISP(5)
      ISP(10)=ISP(9)
      ISP(4)=ISP(5)
      ISP(8)=ISP(9)
C
C      WRITE(*,*)'INPUT THRUST (N) FOR (1), (2), (3)/(5), (6)/(7)'
C      WRITE(*,*)'NOTE: ENTER 0 FOR PHASE 2 AS NECESSARY.        '
      READ(4,*)TH(2),TH(3),TH(5),TH(9)
      TH(6)=50.
      TH(10)=TH(9)
      TH(4)=TH(5)
      TH(8)=TH(9)
      TH(7)=TH(5)
C
C
C      WRITE(*,*)'UPPER STAGE #1 AND #2? (1=LIQUID 2=SOLID 0=NO STAGE)'
      READ(4,*)TUS1,TUS2
      READ(4,*)MFR(2),MFR(3)
```

```
      READ(4,*)MFRLAN,MFRSRC
C
C     ***** MISCELLANEOUS VARIABLE INITIALIZATION *****
C
      MFDV(11)=0.
      TOTDV=0.0
      TOTISP=0.0
      TPROP=0.0
      READ(4,*)MSAMP
      MSRC=MSRC+MSAMP
      H=10
      L=0
      G=9.81
      RPT(3)=0.
      RFT(3)=0.
      ROT(3)=0.
      THPT(3)=0.
      THFT(3)=0.
      THOT(3)=0.
      MPT(3)=0.
      MFT(3)=0.
      MOT(3)=0.
      MCLDFT(3)=0.
      MCLDOT(3)=0.
      EFFFTH(3)=0.
      EFFOTH(3)=0.
C
C     ***** OXYDIZER AND FUEL PARAMETERS *****
C
      R=3.98
      RHOH2O=1000.
      SGFUEL=.437
      SGOXYD=1.521
C
C     ***** PROPELLANT AND PRESSURANT TANK PARAMETERS *****
C
      READ(4,*)PTNK(1),PTNK(2),PTNK(3)
      RHOTNK=4500.0
      RHOCLD=2150.0
      READ(4,*)KPRESS,RPRESS,TO
      SIGCLD=24.821E+6
      SIGCLD=SIGCLD/1.2
      SIGY=400E+6
      UTS=500E+6
      IF((UTS/3.).LT.(SIGY/1.15))THEN
            SIGY=SIGY/1.15
      ELSE
            SIGY=UTS/3.
      ENDIF
C
C      **************************
C      *                        *
C      *  BEGIN MAJOR HANDLING  *
C      *                        *
C      **************************
C
C      WRITE(*,*)'DO YOU WANT A SEPARATE THIRD STAGE FOR TRANSFER'
C      WRITE(*,*)'BACK TO EARTH? (1=LIQUID 2=SOLID 0=NO THIRD STAGE) '
```

```
      READ(4,*)TUS3
      WRITE(3,2)TUS1,TUS2,TUS3
 2    FORMAT('UPPER STAGES, 1=LIQUID 2=SOLID 0=NONE     #1=',F3.0,2X,
     +'#2=',F3.0,2X,'#3=',F3.0)
C
C     ***** SRC TANKS ONLY FOR MIDCOURSE CORRECTION AND EARTH INSERTION *****
C
      IF(TUS3.NE.0)THEN
      TNKSRC=(EXP((DV(8)+DV(9)+DV(10))/(ISP(9)*G))-1.)*
     +MSRC/(1./(1.-MFRSRC)-EXP((DV(8)+DV(9)+DV(10))/
     +(ISP(9)*G)))
      L=8
      READ(4,*)MFR(7)
      GOTO 3
      ENDIF
      L=7
C
C     ***** SRC TANKS FOR ALL DELTA-V'S FROM THE ASTEROID *****
C
      TNKSRC=(EXP((DV(7)+DV(8)+DV(9)+DV(10))/(ISP(9)*G))-1.)*
     +MSRC/(1./(1.-MFRSRC)-EXP((DV(7)+DV(8)+DV(9)+DV(10))/
     +(ISP(9)*G)))
C
C     ***** SRC FUEL MASS REQUIRED FOR EACH BURN *****
C
 3    HAWK1=MSRC+TNKSRC
      DO 5 J=H,L,-1
        MFDV(J)=(EXP(DV(J)/(ISP(9)*G))-1.)*HAWK1
        HAWK1=HAWK1+MFDV(J)
C
C    ***** CALCULATION OF TOTAL IMPULSE AND BURN TIMES *****
C
        ITOT(J)=MFDV(J)*ISP(9)*G
        BT(J)=ITOT(J)/TH(J)
 5    CONTINUE
C
C     ***** FUEL/STRUCTURAL MASS FOR SRC'S JETTISONABLE BOOSTER STAGE *****
C
      IF(L.EQ.8)THEN
        A=(1.-EXP(DV(7)/(ISP(7)*G)))
        B=(EXP(DV(7)/(ISP(7)*G)))*(1./MFR(7)-1.)-1./MFR(7)
        MFDV(7)=(A/B)*HAWK1
        ITOT(7)=MFDV(7)*ISP(7)*G
        BT(7)=ITOT(7)/TH(7)
        TNKSTR(7)=MFDV(7)/MFR(7)-MFDV(7)
        TOTSRC=HAWK1+TNKSTR(7)+MFDV(7)-MSAMP
      ELSE
C
C
C     ***** TOTAL SRC MASS USING NON-JETTISONABLE TANKS FOR ENTIRE *****
C     *****          JOURNEY BACK TO EARTH FROM ASTEROID           *****
C
C
        TOTSRC=MSRC+TNKSRC/(1.-MFRSRC)-MSAMP
      ENDIF
C
C
C     ***** LANDER TANK MASS BASED ON FULLY FUELED SRC AND LANDER MASS *****
```

```
C
C
  10  TNKLAN=((EXP((DV(4)+DV(5))/(ISP(5)*G))-1.)*MORB+(EXP((DV(4)+
     +DV(5)+DV(6))/(ISP(5)*G))-1.)*(TOTSRC+ML))/(1./(1.-MFRLAN)-
     +EXP((DV(4)+DV(5)+DV(6))/(ISP(5)*G)))
C
C     ***** LANDER FUEL MASS REQUIRED FOR EACH BURN *****
C
      MFDV(6)=(EXP(DV(6)/(ISP(5)*G))-1.)*(TOTSRC+TNKLAN+ML)
      MFDV(5)=(EXP(DV(5)/(ISP(5)*G))-1.)*(TOTSRC+TNKLAN+ML+MORB+
     +MFDV(6))
      MFDV(4)=(EXP(DV(4)/(ISP(5)*G))-1.)*(TOTSRC+TNKLAN+ML+MORB+
     +MFDV(6)+MFDV(5))
C

C     ***** CALCULATION OF TOTAL IMPULSE AND BURN TIMES *****
C
      DO 11 J=4,6
        ITOT(J)=MFDV(J)*ISP(5)*G
        BT(J)=ITOT(J)/TH(J)
 11   CONTINUE
      TOTLAN=ML+TNKLAN/(1.-MFRLAN)
C
C
C     ***** LEO-TO-ASTEROID UPPER STAGE(S) TANK AND PROPELLANT MASS *****
C
C
      HAWK=TOTSRC+TOTLAN+MORB
 13   DO 15 K=3,2,-1
        IF((K.EQ.3).AND.(ISP(K).EQ.0.0))THEN
        MFDV(K)=0.0
        TNKSTR(K)=0.0
        GOTO 14
        ENDIF
        A=(1.-EXP(DV(K)/(ISP(K)*G)))
        B=(EXP(DV(K)/(ISP(K)*G)))*(1./MFR(K)-1.)-1./MFR(K)
        MFDV(K)=(A/B)*HAWK
        ITOT(K)=MFDV(K)*ISP(K)*G
        BT(K)=ITOT(K)/TH(K)
        TNKSTR(K)=MFDV(K)/MFR(K)-MFDV(K)
 14     HAWK=HAWK+TNKSTR(K)+MFDV(K)
 15   CONTINUE
C
C
C     ***** MASS OF THE HAWKING BEFORE EVERY PRIMARY BURN *****
C
C
      MCOMP(10)=MSRC+TNKSRC+MFDV(10)
      MCOMP(9)=MCOMP(10)+MFDV(9)
      MCOMP(8)=MCOMP(9)+MFDV(8)
      MCOMP(7)=TOTSRC+MSAMP
      MCOMP(6)=MCOMP(7)+ML+TNKLAN+MFDV(6)-MSAMP
      MCOMP(5)=MCOMP(6)+MFDV(5)+MORB
      MCOMP(4)=MCOMP(5)+MFDV(4)
      MCOMP(3)=MCOMP(4)+MFDV(3)+TNKSTR(3)
      MCOMP(2)=MCOMP(3)+MFDV(2)+TNKSTR(2)
C
C
```

```
C      ***** CALCULATION OF TOTAL PROPELLANT, IMPULSE, AND DELTA-V *****
C      *****                  FOR THE ENTIRE MISSION                 *****
C
C
       DO 16 J=2,10
         TPROP=TPROP+MFDV(J)
         TOTISP=TOTISP+ITOT(J)
         TOTDV=TOTDV+DV(J)
 16    CONTINUE
C
C




C      *****************************
C      *                           *
C      * PROPELLANT AND PRESSURANT *
C      *        TANK SIZING        *
C      *                           *
C      *****************************
C
C      ***** VOLUME & MASS OF LANDER FUEL, OXYDIZER, & PRESSURANT *****
C
       VF(1)=1.05*((MFDV(4)+MFDV(5)+MFDV(6))/(RHOH2O*SGFUEL))*(1/(R+1))
       MF(1)=RHOH2O*SGFUEL*VF(1)
       VO(1)=1.05*((MFDV(4)+MFDV(5)+MFDV(6))/(RHOH2O*SGOXYD))*(R/(R+1))
       MO(1)=RHOH2O*SGOXYD*VO(1)
       MPO(1)=((PTNK(1)*VO(1))/(RPRESS*TO))*1.05*(KPRESS/
      +(1-(PTNK(1)/PTNK(3))))
       MPF(1)=((PTNK(1)*VF(1))/(RPRESS*TO))*1.05*(KPRESS/
      +(1-(PTNK(1)/PTNK(3))))
       VPO(1)=1.05*MPO(1)*RPRESS*TO/PTNK(3)
       VPF(1)=1.05*MPF(1)*RPRESS*TO/PTNK(3)
C
C
C      ***** VOLUME & MASS OF SRC FUEL, OXYDIZER, & PRESSURANT *****
C      *****           FOR NON-JETTISONABLE SRC TANKS          *****
C
C
       IF(TUS3.EQ.0)THEN
       VF(2)=1.05*((MFDV(10)+MFDV(9)+MFDV(8)+MFDV(7))/(RHOH2O*SGFUEL))
      +*(1/(R+1))
       MF(2)=RHOH2O*SGFUEL*VF(2)
       VO(2)=1.05*((MFDV(10)+MFDV(9)+MFDV(8)+MFDV(7))/(RHOH2O*SGOXYD))
      +*(R/(R+1))
       MO(2)=RHOH2O*SGOXYD*VO(2)
       MPO(2)=((PTNK(1)*VO(2))/(RPRESS*TO))*1.05*(KPRESS/
      +(1-(PTNK(1)/PTNK(3))))
       MPF(2)=((PTNK(1)*VF(2))/(RPRESS*TO))*1.05*(KPRESS/
      +(1-(PTNK(1)/PTNK(3))))
       VPO(2)=1.05*MPO(2)*RPRESS*TO/PTNK(3)
       VPF(2)=1.05*MPF(2)*RPRESS*TO/PTNK(3)
       N=2
       GOTO 17
       ENDIF
C
C
```

```
C      ***** VOLUME & MASS OF SRC FUEL, OXYDIZER, & PRESSURANT *****
C      *****    FOR MIDCOURSE CORRECTION AND EARTH INSERTION    *****
C
C
       VF(2)=1.05*((MFDV(10)+MFDV(9)+MFDV(8))/(RHOH2O*SGFUEL))*(1/(R+1))
       MF(2)=RHOH2O*SGFUEL*VF(2)
       VO(2)=1.05*((MFDV(10)+MFDV(9)+MFDV(8))/(RHOH2O*SGOXYD))*(R/(R+1))
       MO(2)=RHOH2O*SGOXYD*VO(2)
       MPO(2)=((PTNK(1)*VO(2))/(RPRESS*TO))*1.05*(KPRESS/
      +(1-(PTNK(1)/PTNK(3))))
       MPF(2)=((PTNK(1)*VF(2))/(RPRESS*TO))*1.05*(KPRESS/
      +(1-(PTNK(1)/PTNK(3))))
       VPO(2)=1.05*MPO(2)*RPRESS*TO/PTNK(3)
       VPF(2)=1.05*MPF(2)*RPRESS*TO/PTNK(3)
C
C
C      ***** VOLUME & MASS OF FUEL, OXYDIZER AND PRESSURANT *****
C      *****            FOR SRC'S JETTISONABLE BOOSTER       *****
C
C
       VF(3)=1.05*(MFDV(7)/(RHOH2O*SGFUEL))*(1/(R+1))
       MF(3)=RHOH2O*SGFUEL*VF(3)
       VO(3)=1.05*(MFDV(7)/(RHOH2O*SGOXYD))*(R/(R+1))
       MO(3)=RHOH2O*SGOXYD*VO(3)
       MPO(3)=((PTNK(1)*VO(3))/(RPRESS*TO))*1.05*(KPRESS/
      +(1-(PTNK(1)/PTNK(3))))
       MPF(3)=((PTNK(1)*VF(3))/(RPRESS*TO))*1.05*(KPRESS/
      +(1-(PTNK(1)/PTNK(3))))
       VPO(3)=1.05*MPO(3)*RPRESS*TO/PTNK(3)
       VPF(3)=1.05*MPF(3)*RPRESS*TO/PTNK(3)
       N=3
C
C
C      ***** CALCULATE TANK MASS, RADIUS, AND THICKNESS *****
C      *****        FOR SRC, LANDER, AND SRC BOOSTER     *****
C
 17    DO 18 J=1,N
        RPT(J)=((3.*(VPO(J)+VPF(J)))/(4.*ACOS(-1.)))**(1./3.)
        RFPT(J)=((3.*VPF(J))/(4.*ACOS(-1.)))**(1./3.)
        ROPT(J)=((3.*VPO(J))/(4.*ACOS(-1.)))**(1./3.)
        RFT(J)=((3.*VF(J))/(4.*ACOS(-1.)))**(1./3.)
        ROT(J)=((3.*VO(J))/(4.*ACOS(-1.)))**(1./3.)
        THPT(J)=PTNK(3)*RPT(J)/(2.*SIGY)
        THFPT(J)=PTNK(3)*RFPT(J)/(2.*SIGY)
        THOPT(J)=PTNK(3)*ROPT(J)/(2.*SIGY)
        THFT(J)=(PTNK(1)*RFT(J)/(2.*SIGY))/(1.-.7*PTNK(1)/(2.*SIGY))
        THOT(J)=(PTNK(2)*ROT(J)/(2.*SIGY))/(1.-.7*PTNK(2)/(2.*SIGY))
        MPT(J)=(4.*RHOTNK*ACOS(-1.)/3.)*((RPT(J)+THPT(J)/2.)**3
      + -(RPT(J)-THPT(J)/2.)**3)
        MFPT(J)=(4.*RHOTNK*ACOS(-1.)/3.)*((RFPT(J)+THFPT(J)/2.)**3
      + -(RFPT(J)-THFPT(J)/2.)**3)
        MOPT(J)=(4.*RHOTNK*ACOS(-1.)/3.)*((ROPT(J)+THOPT(J)/2.)**3
      + -(ROPT(J)-THOPT(J)/2.)**3)
        MFT(J)=(4.*RHOTNK*ACOS(-1.)/3.)*((RFT(J)+1.7*THFT(J))**3
      + -(RFT(J)+.7*THFT(J))**3)
        MOT(J)=(4.*RHOTNK*ACOS(-1.)/3.)*((ROT(J)+1.7*THOT(J))**3
      + -(ROT(J)+.7*THOT(J))**3)
        MCLDFT(J)=(4.*RHOCLD*ACOS(-1.)/3.)*((RFT(J)+.7*THFT(J))**3-
```

```
     + (RFT(J))**3)
       MCLDOT(J)=(4.*RHOCLD*ACOS(-1.)/3.)*((ROT(J)+.7*THOT(J))**3-
     + (ROT(J))**3)
       MNEWFT(J)=MCLDFT(J)+MFT(J)
       MNEWOT(J)=MCLDOT(J)+MOT(J)
       MNEW(J)=MNEWFT(J)+MNEWOT(J)
       THCLDF(J)=.7*THFT(J)
       THCLDO(J)=.7*THOT(J)
       EFFFTH(J)=THFT(J)+(SIGCLD/SIGY)*(.7*THFT(J)-.1*THFT(J))
       EFFOTH(J)=THOT(J)+(SIGCLD/SIGY)*(.7*THOT(J)-.1*THOT(J))
 18    CONTINUE
C
C


C      **********************
C      *                    *
C      *  FORMATTED RESULTS *
C      *                    *
C      **********************
C
C
 21    WRITE(3,22)MSRC-MSAMP,ML,MORB
 22    FORMAT('HAWKING DRY MASSES:     SRC = ',F5.1,5X,'LANDER = ',F5.1
     +,5X,'ORBITER = ',F5.1)
       WRITE(3,*)
       WRITE(3,*)'PHASE DELTA  ISP    BURN     FUEL     FUEL     TOTAL   '
     +,'    MASS     HAWKING'
       WRITE(3,*)'           V          TIME     MASS     FRAC.    IMPULSE '
     +,'    RATIO    MASS'
       WRITE(3,*)'        (m/s)  (s)     (s)      (kg)              (N-s)   '
     +,'    (kg)      (kg)'
       WRITE(3,*)'--------------------------------------------------------'
     +,'-----------------'
       WRITE(3,51)1,DV(2),ISP(2),BT(2),MFDV(2),MFR(2),ITOT(2),MCOMP(2)
       WRITE(3,48)2,DV(3),ISP(3),BT(3),MFDV(3),MFR(3),ITOT(3),
     +MCOMP(3)/MCOMP(2),MCOMP(3)
       IF(TUS2.EQ.0)THEN
       MCOMP(3)=MCOMP(2)
       ENDIF
       WRITE(3,49)DV(4),ISP(4),BT(4),MFDV(4),MFRLAN,ITOT(4),
     +MCOMP(4)/MCOMP(3),MCOMP(4)
       WRITE(3,48)3,DV(5),ISP(5),BT(5),MFDV(5),(MFDV(5)+MFDV(6))/
     +(TNKLAN/(1.-MFRLAN)),ITOT(5),MCOMP(5)/MCOMP(4),MCOMP(5)
       WRITE(3,48)4,DV(6),ISP(6),BT(6),MFDV(6),MFDV(6)/
     +(TNKLAN/(1.-MFRLAN)),ITOT(6),MCOMP(6)/MCOMP(5),MCOMP(6)
       WRITE(3,48)5,DV(7),ISP(7),BT(7),MFDV(7),MFDV(7)/
     +(TNKSTR(7)+MFDV(7)),ITOT(7),MCOMP(7)/MCOMP(6),MCOMP(7)
       WRITE(3,49)DV(8),ISP(8),BT(8),MFDV(8),MFRSRC,ITOT(8),
     +MCOMP(8)/MCOMP(7),MCOMP(8)
       DO 47 J=9,10
       WRITE(3,48)J-3,DV(J),ISP(J),BT(J),MFDV(J),(MFDV(J)+MFDV(J+1))/
     +(TNKSRC/(1.-MFRSRC)),ITOT(J),MCOMP(J)/MCOMP(J-1),MCOMP(J)
 47    CONTINUE
 48    FORMAT(I2,4X,F6.0,2X,F4.0,2X,F5.0,2X,F8.0,2X,F5.4,F12.0,2X,
     +F5.4,3X,F8.0)
 49    FORMAT('MCB',3X,F6.0,2X,F4.0,2X,F5.0,2X,F8.0,2X,F5.4,F12.0,2X,
     +F5.4,3X,F8.0)
```

```
      WRITE(3,*)'---------------------------------------------------'
     +,'-----------------------'
      WRITE(3,50)TOTDV,TPROP,TOTISP,MCOMP(10)/MCOMP(2)
 50   FORMAT('TOTAL',F7.0,16X,F7.0,7X,F12.0,2X,F5.4)
 51   FORMAT(I2,4X,F6.0,2X,F4.0,2X,F5.0,2X,F8.0,2X,F5.4,F12.0,3X,
     +'----',3X,F8.0)
      WRITE(3,*)
      WRITE(3,*)'SRC MASS (kg):'
      WRITE(3,*)'  BEFORE MAIN TRANSFER BURN = ',MCOMP(7)
      WRITE(3,*)'  BEFORE EARTH RENDEZVOUS   = ',MCOMP(9)
      WRITE(3,*)'  BEFORE EARTH INSERTION    = ',MCOMP(10)
      WRITE(3,*)'SRC FUEL TANK MASS          = ',TNKSRC
      WRITE(3,*)
      WRITE(3,*)'LANDER MASS (kg):'
      WRITE(3,*)'  BEFORE ASTEROID INSERTION = ',ML+TNKLAN+MFDV(5)+
     +MFDV(6)
      WRITE(3,*)'  BEFORE SURFACE RENDEZVOUS = ',ML+TNKLAN+MFDV(6)
      WRITE(3,*)'  AFTER SURFACE RENDEZVOUS  = ',ML+TNKLAN
      WRITE(3,*)'LANDER FUEL TANK MASS       = ',TNKLAN
      WRITE(3,*)
      WRITE(3,*)'MASS OF UPPER STAGES (kg): PROPELLANT        TANKS'
      WRITE(3,55)MFDV(2),TNKSTR(2)
      WRITE(3,60)MFDV(3),TNKSTR(3)
      IF(TUS3.NE.0)THEN
      WRITE(3,54)MFDV(7),TNKSTR(7)
 54   FORMAT('  ASTEROID-> EARTH #3 = ',3X,F9.2,9X,F9.2)
      ENDIF
 55   FORMAT('  LEO TO ASTEROID, #1 = ',3X,F9.2,9X,F9.2)
 60   FORMAT('  LEO TO ASTEROID, #2 = ',3X,F9.2,9X,F9.2)
      WRITE(3,*)
      WRITE(3,*)'             PROPELLANT/PRESSURANT MASS (kg) & ',
     +'VOLUME (m^3)'
      WRITE(3,*)'                                                   '
      WRITE(3,*)'PROPELLANT              LANDER           UPPR.STG. #3',
     +'           SRC  '
      WRITE(3,*)'                        V      M             V      M   ',
     +'           V       M'
      WRITE(3,*)'---------------------------------------------------',
     +'-------------------------'
      WRITE(3,65)VF(1),MF(1),VF(3),MF(3),VF(2),MF(2)
      WRITE(3,70)VO(1),MO(1),VO(3),MO(3),VO(2),MO(2)
      WRITE(3,75)VPO(1)+VPF(1),MPO(1)+MPF(1),VPO(3)+VPF(3),MPO(3)+
     +MPF(3),VPO(2)+VPF(2),MPO(2)+MPF(2)
 65   FORMAT('FUEL',15X,F6.4,2X,F6.1,6X,F6.4,2X,F6.1,6X,F6.4,2X,F6.1)
 70   FORMAT('OXYDIZER',11X,F6.4,2X,F6.1,6X,F6.4,2X,F6.1,6X,F6.4,
     +2X,F6.1)
 75   FORMAT('PRESSURANT',9X,F6.4,2X,F6.1,6X,F6.4,2X,F6.1,6X,F6.4,
     +2X,F6.1)
      WRITE(3,*)
      WRITE(3,*)'          PROPELLANT & PRESSURANT TANK MASSES (kg) ',
     +'& DIMENSIONS (m)'
      WRITE(3,*)
      WRITE(3,*)'# OF          LANDER                   UPPR.STG. #3
     +         SRC  '
      WRITE(3,*)'TANKS      R       T        M         R         T        M
     +   R        T        M'
      WRITE(3,*)'---------------------------------------------------------
     +----------------------'
```

```
      WRITE(3,90)RFT(1),THFT(1),MFT(1),RFT(3),THFT(3),MFT(3),RFT(2),
     +THFT(2),MFT(2)
      WRITE(3,98)THCLDF(1),MCLDFT(1),THCLDF(3),MCLDFT(3),
     +THCLDF(2),MCLDFT(2)
      WRITE(3,101)EFFFTH(1),EFFFTH(3),EFFFTH(2)
      WRITE(3,*)
      WRITE(3,95)ROT(1),THOT(1),MOT(1),ROT(3),THOT(3),MOT(3),ROT(2),
     +THOT(2),MOT(2)
      WRITE(3,99)THCLDO(1),MCLDOT(1),THCLDO(3),MCLDOT(3),
     +THCLDO(2),MCLDOT(2)
      WRITE(3,101)EFFOTH(1),EFFOTH(3),EFFOTH(2)
      WRITE(3,100)MNEW(1),MNEW(3),MNEW(2)
      WRITE(3,*)
      WRITE(3,80)RPT(1),THPT(1),MPT(1),RPT(3),THPT(3),MPT(3),RPT(2),
     +THPT(2),MPT(2)
      WRITE(3,85)RFPT(1),THFPT(1),MFPT(1),RFPT(3),THFPT(3),MFPT(3),
     +RFPT(2),THFPT(2),MFPT(2)
      WRITE(3,87)ROPT(1),THOPT(1),MOPT(1),ROPT(3),THOPT(3),MOPT(3),
     +ROPT(2),THOPT(2),MOPT(2)
      WRITE(3,*)
      WRITE(3,100)MNEW(1)+MPT(1),MNEW(3)+MPT(3),MNEW(2)+MPT(2)
      WRITE(3,100)MNEW(1)+MFPT(1)+MOPT(1),MNEW(3)+MFPT(3)+MOPT(3),
     +MNEW(2)+MFPT(2)+MOPT(2)
 80   FORMAT('FOPTNK',1X,F5.2,1X,E9.4,F7.2,2X,F5.2,1X,
     +E9.4,2X,F6.2,2X,F5.2,2X,E9.4,F6.2)
 85   FORMAT('F PTNK',1X,F5.2,1X,E9.4,F7.2,2X,F5.2,1X,
     +E9.4,2X,F6.2,2X,F5.2,2X,E9.4,F6.2)
 87   FORMAT('O PTNK',1X,F5.2,1X,E9.4,F7.2,2X,F5.2,1X,
     +E9.4,2X,F6.2,2X,F5.2,2X,E9.4,F6.2)
 90   FORMAT('1 FUEL',1X,F5.2,1X,E9.4,F7.2,2X,F5.2,2X,
     +E9.4,1X,F6.2,2X,F5.2,2X,E9.4,F6.2)
 95   FORMAT('1 OXYD',1X,F5.2,1X,E9.4,F7.2,2X,F5.2,2X,
     +E9.4,1X,F6.2,2X,F5.2,2X,E9.4,F6.2)
 98   FORMAT('TNK CLAD',5X,E9.4,F7.2,9X,E9.4,F7.2,9X,
     +E9.4,F6.2)
 99   FORMAT('TNK CLAD',5X,E9.4,F7.2,9X,E9.4,F7.2,9X,
     +E9.4,F6.2)
100   FORMAT('TOTALS',16X,F7.2,18X,F7.2,17X,F7.2)
101   FORMAT('E.T.',9X,E9.4,16X,E9.4,16X,E9.4)
      STOP
      END
```

Appendix D: Communication Link Design

Orbiter HGA Link Design Control Table

Transmitter Parameters:	
RF power to antenna, dBm	13.01
Antenna Gain, dBi	49.07
Pointing Loss, dB	0.00
Total System Noise Temp, K	560.00
Path Parameters:	
Space Loss, dB	-284.01
Frequency = 8420.43 MHz	
Range = 4.500+08 km	
= 3.0083 AU	
Atmospheric Attenuation, dB	-0.10
Line Loss, dB	-3.00
Signal-to-Noise Ratio	9.00
Receiver Parameters	
Polarization Loss, dB	00.00
DSN Antenna Gain, dBi	73.94
Pointing Loss, dB	00.00
Data Channel Performance	
Data Bit Rate, dB	41.03
Range = 3.0083 AU	
Bit Rate = 12680.9 bps	

Orbiter/Lander LGA Link Design Control

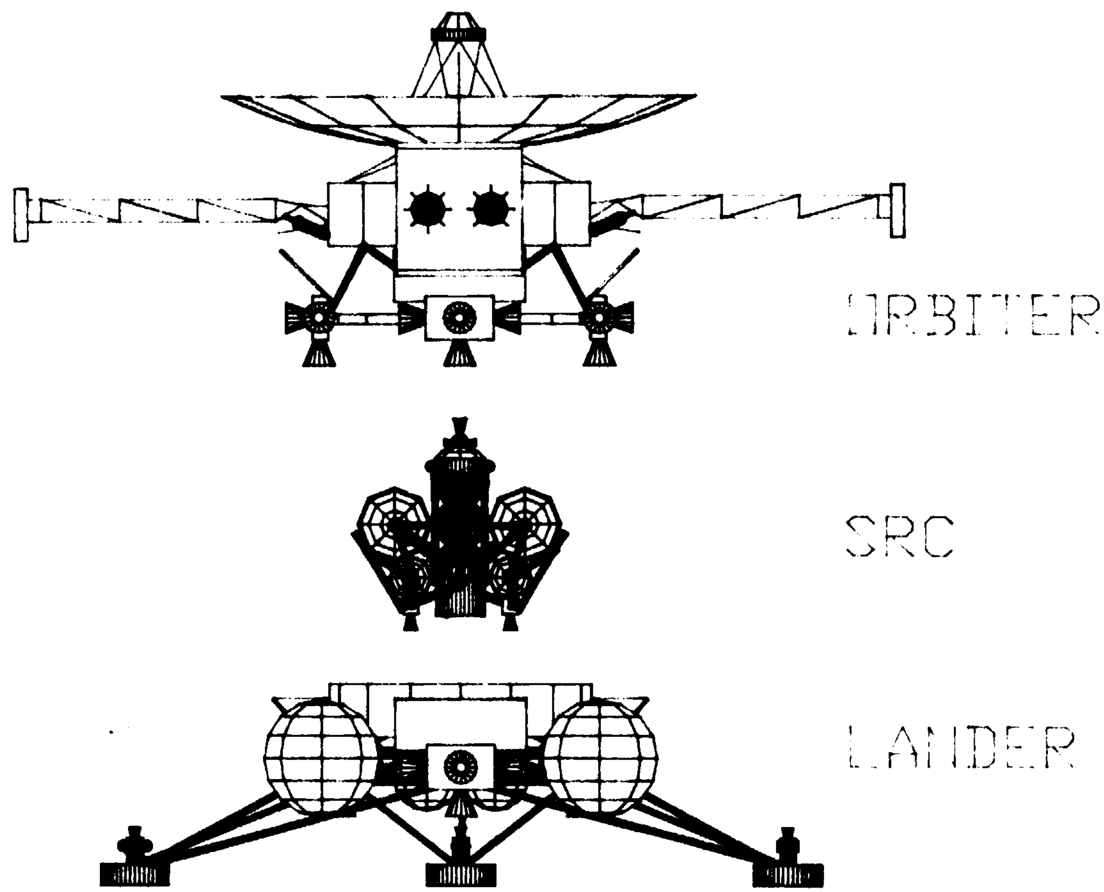
Transmitter Parameters:	
RF power to antenna, dBm	6.99
Antenna Gain, dBi	16.27
Pointing Loss, dB	0.00
Total System Noise Temp, K	900.00
Path Parameters:	
Space Loss, dB	-132.92
Frequency = 2114.68MHz	
Range = 50.0 km	
= 3.342-7 AU	
Atmospheric Attenuation, dB	00.00
Line Loss, dB	-1.00
Signal-to-Noise Ratio	65.00
Receiver Parameters	
Polarization Loss, dB	00.00
DSN Antenna Gain, dBi	73.94
Pointing Loss, dB	00.00
Data Channel Performance	
Data Bit Rate, dB	51.35
Range = 50.0 km	
Bit Rate = 136,440 bps	

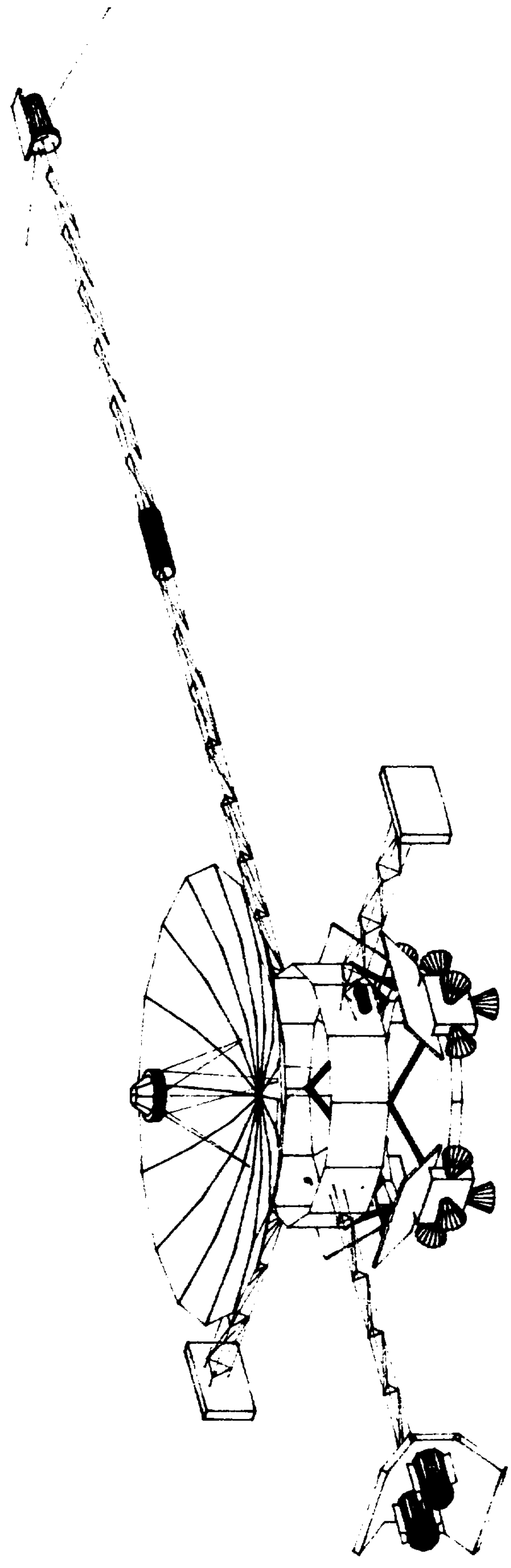
SRC LGA Link Design Control Table

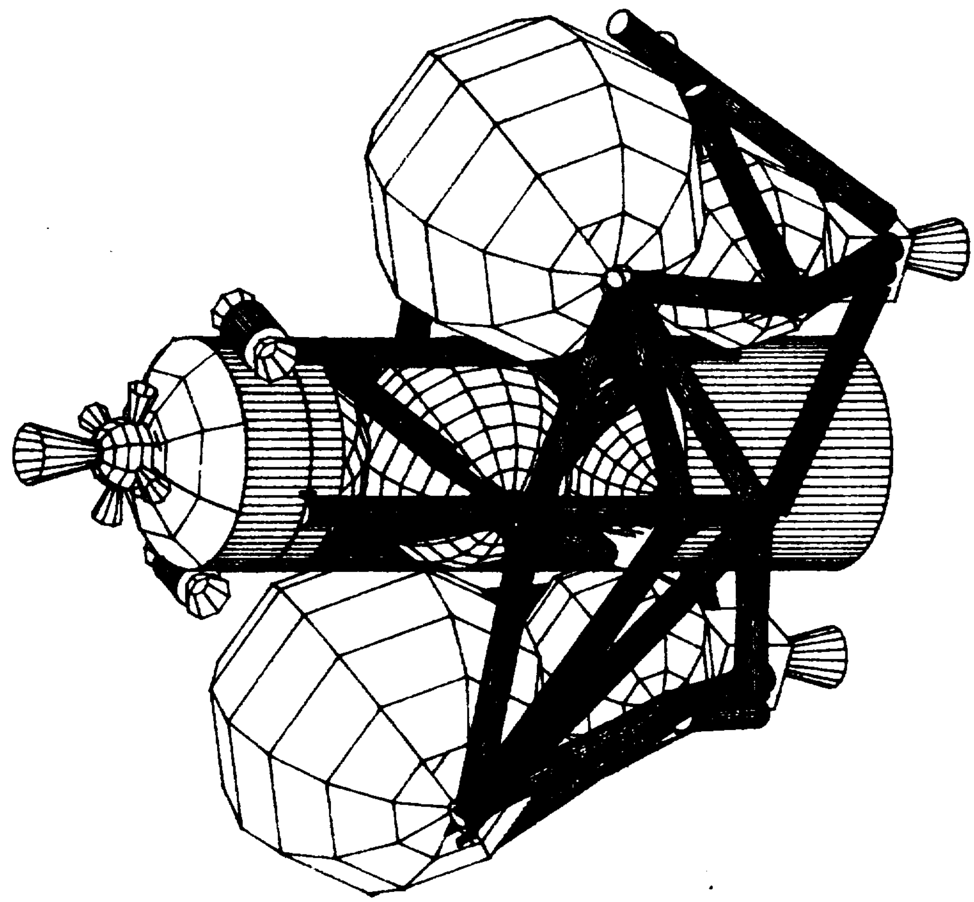
Transmitter Parameters:	
RF power to antenna, dBm	6.99
Antenna Gain, dBi	28.06
Pointing Loss, dB	0.00
Total System Noise Temp, K	560.00
Path Parameters:	
Space Loss, dB	-284.03
Frequency = 8435.14 MHz	
Range = 4.500+08 km	
= 3.0083 AU	
Atmospheric Attenuation, dB	-0.10
Line Loss, dB	-3.00
Signal-to-Noise Ratio	6.00
Receiver Parameters	
Polarization Loss, dB	00.00
DSN Antenna Gain, dBi	73.94
Pointing Loss, dB	00.00
Data Channel Performance	
Data Bit Rate, dB	16.99
Range = 3.0083 AU	
Bit Rate = 50.00 bps	

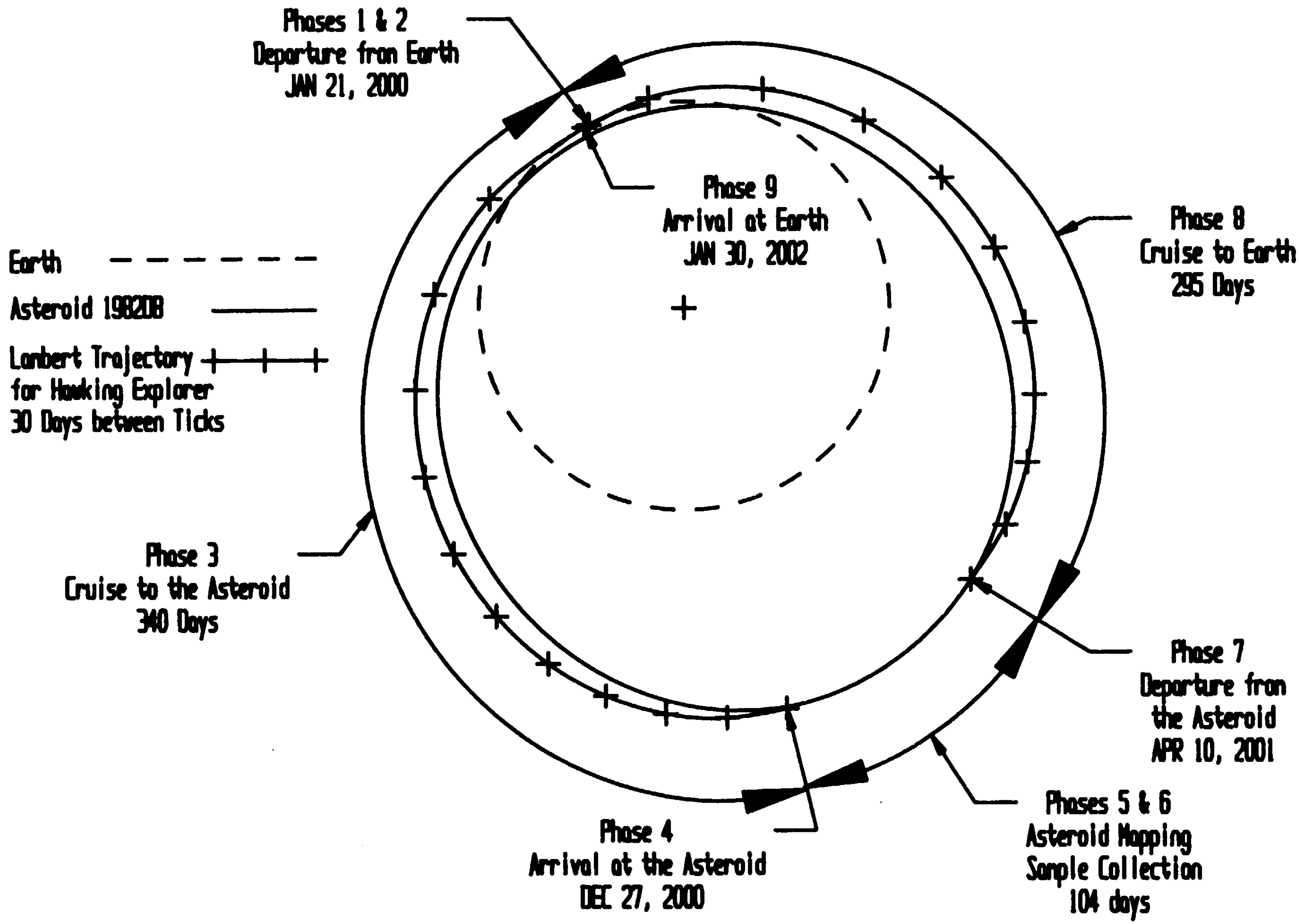
Appendix E: CAD Drawings

HAWKING EXPLORER









Phases 1 & 2
Departure from Earth
JAN 21, 2000
Phase 9
Arrival at Earth
JAN 30, 2002
Phase 8
Cruise to Earth
295 Days
Earth
Asteroid 1982DB
Lambert Trajectory
for Hawking Explorer
30 Days between Ticks
Phase 3
Cruise to the Asteroid
340 Days
Phase 7
Departure from
the Asteroid
APR 10, 2001
Phases 5 & 6
Asteroid Mapping
Sample Collection
104 days
Phase 4
Arrival at the Asteroid
DEC 27, 2000